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ADVISORY GROUP FOR AEROSPACE RESEARCH & DEVELOPMENT

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The Propulsion and Energetics Panel wishes to express its thanks to the hosts, the Dutch National Delegates to AGARD, for the invitation to hold its 49th Meeting at the Koninklijk Instituut van Ingenieurs in The Hague, The Netherlands, and for the provision of the necessary facilities and personnel to make this meeting possible.

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TECHNICAL EVALUATION REPORT

by

G.P.Sallee

1. INTRODUCTION

The 49th meeting of the Propulsion and Energetics Panel on Power Plant Reliability was held at the Koninklijk Instituut Van Ingenieurs, The Hague, Netherlands on 31 March and 1 April 1977. The meeting was organized to review and discuss engine reliability from four aspects:

- the reliability of current civil and military engines as experienced by the users,
- civil and military authorities' plans to promote improved reliability in future engines,
- what manufacturers are doing to improve reliability through design and testing programs,
- the role that engine health monitoring and diagnostics is taking in minimizing the impact of engine unreliability for both civil and military users.

The meeting was divided into four sessions with a total of 18 papers. The presentations were well received and the meeting was beneficial in establishing a basis for future discussions.

2. SUMMARY

The following observations reflect the tone of the meeting and the major results.

- Engine reliability is not satisfactory in either commercial or military services. In particular the newer commercial engines are not living up to operators expections.
- It seems that civil and military authorities are considering the promulgation of more stringent requirements and standards concerning the development, certification/qualification and acquisition of future engines with respect to the reliability requirements that must be met.
- Manufacturers are designing for improved maintainability and employing improved testing techniques to expose problems early. Further progress is possible, but is contingent on the availability of engineering data on actual engine usage in military service. In addition detailed part failure data is needed to determine the causes for part failure with respect to usage and the relationships that exist between the various modes of failure.
- The economic impact of military engine unreliability has not been discussed. The cost consequences of premature engine removals, aborts, part failures, etc., are needed to establish the role of engine reliability in engine life cycle cost.
- The growth of engine health monitoring in the commercial airlines and the increased experimentation of such approaches in the military are indicative of the serious consequences of poor engine reliability. The future growth/potential for such techniques is impressive.

3. RECOMMENDATIONS

Engine reliability will continue to increase in importance in the light of current economic pressures. The following recommendations are believed by the writer to be worthy of consideration as items for further research.

- The reliability history of todays engines beginning with their initial entry into service needs to be examined such that trends and major causes of unreliability are visible to both designers and users. Comparison of available data between users suggests that major differences exist in causes for engine removals on identical airframe/engine combinations. Concurrent with examining historical reliability data is the need to quantify the cost of engine maintenance. The availability of such data would permit proper assessment of the trade-offs between performance improvements and reliability improvements for future designs.
- Continued documentation on the manner in which engines are actually used in service is needed. The data currently available from recent testing show that actual thermal cycles during flight are more numerous and different than contemplated by design specifications or qualification test programs.
- Broader adaptation and experimentation in engine health monitoring techniques is recommended for the military services. Preliminary estimates on military engine maintenance cost indicate that they are four to ten times higher than levels previously reported. When fully documented the cost of maintenance will undoubtably support strong action for cost reductions, and health monitoring has the potential for providing the cost savings from controlling unnecessary maintenance actions and reducing the costs of failures.

- Continued development and application of accelerated engine testing techniques and lead-the-fleet engine concepts are recommended. In addition there is a need to more closely simulate the installed engine environment. Externally applied aerodynamic, gravitational and gyroscopic loads can cause ovalization of cases and local seal rubouts. The resulting losses in performance are important to the life of critical hot section parts.

- Examination of the role of subsystem reliability in overall propulsion system reliability is needed. The cost consequences of flight delays, cancellations or mission aborts to both military and commercial engine users need to be defined and exposed to the technical community.

- The military services, like the commercial airlines, must insure that the consequences of missing reliability goals are similiar to those for missing performance guarantees. In order for this to occur more definitive research will be needed to support the establishment of reasonable requirements in keeping with the overall demands placed on future engines.

- Regarding engine health monitoring the military use of commercial airline procedures should be initiated slowly in an evolutionary rather than revolutionary fashion. The airlines have more engineering personnel per installed engine than the military services which permits them to monitor trends on an individual engine basis and take corrective action quickly. The availability of trained personnel is critical in the successful application of such techniques.

4. DISCUSSION

In preparing this evaluation report, the papers presented were regrouped to provide a more harmonious presentation on each of the four aspects of engine reliability. The first aspect to be reviewed was a status report on civil and military engine reliability and concerns related thereto. In listening and reviewing these papers and the discussions that followed the impact of poor reliability on safety and life cycle costs were stressed.

Session I

Paper No. 6 by J.A.Aguer discusses problems being experienced in todays high bypass ratio commercial engines and the origin of major structural and fire hazards. Foreign object damage, titanium fires, coking of fuel nozzle passages, bearing sump fires and bearing failures are typical of such problems. Maintenance can not correct such deficiencies and improvement in design standards to avoid such problems must be sought,

Historically, engine weight is increased in early service through modifications to improve reliability and durability and at significant cost to the users. More emphasis is needed on structural integrity and durability in engine development programs and less on weight reduction.

Paper No.5 by S.K.W.J.Demarteau continues this discussion and provides an understanding of the cost consequences of poor engine reliability to commercial airlines. As an example, an engine removal at an airport close to the main maintenance base is 1/10 the cost of a removal at a distant airport overseas. Early detection of failures is also essential to control the high cost of secondary damage. Low time failures and removals of recently repaired high bypass engines are a major concern today and are slowing progress toward attainment of desired reliability goals. The only way to achieve a justified cost/reliability level is to look for cost effectiveness in modifications, maintenance and monitoring practices.

Paper No. 2 by Gen. L.Giorgeri and Col. G.Facca discusses the broad aspects of reliability and related cost in Military Air Forces. Engines have a tendency to lose performance with the passage of time in service and this loss is not fully recovered during maintenance. Many of the causes for engine removals are common for all nations; however, many unique items exist in each nation for the same engine and aircraft combination. The cost of engine maintenance in the military services is much higher than previously believed. These factors lead to concern about future engines of increased complexity. Additional detailed studies are needed to understand the overall reliability picture of today's engines.

These papers summarized the status of current engine reliability. The reliability of high bypass ratio engines entering commercial service are still less than what is needed. These new engines, even after five to seven years and millions of hours of service, are still poorer than their predecessors. The military experience, while not alarming, is less than satisfactory. A study of airline experience with their newer high performance engines brings serious concern about the initial reliability of future military engines. Consideration of changing military design objectives and contract incentives to favor improved reliability even at the expense of some performance losses may be needed to ensure adequate readiness and reasonable life-cycle-costs.

Session II

Civil and military authorities appear to be reacting to this perceived state of affairs. Mr. J.Slatford presents some of his thoughts concerning future civil engine reliability requirements in Paper No. 4 and the form that such requirements might take. The changes in engine development and procurement program policies being contemplated by the

United States Air Force are discussed in Paper No. 1. The revised program policies stress more steps in the process going from original concept to actual full scale production, and increased emphasis on demonstration testing for structural integrity and reliability. More decision points and reviews would be undertaken prior to full production release. The paper by R.Holl (No. 7) discusses the evaluation of general engine specifications and type test requirements. He points out that little effort was made until recently to obtain meaningful data of how engines were actually used in military service. Problems with engine reliability and establishment of appropriate engine part life standards led to development of equipment to gather data on how engines were operated in service. The results of these efforts show actual usage to be far more severe than estimated and signal that a rather severe change is needed in engine design, development testing, maintenance and reliability criteria. Further work in correlating parts life consumption based on actual speed and temperature excursions and their variation with flight type is planned and will be beneficial in establishing improved part life standards.

Paper No. 11 by R.J.Hill discusses procedural steps for predicting the life of turbine components. The difficulties in establishing good design and life positions are increased when actual usage varies in an unknown manner from design duty cycle assumptions. The ability to determine the proper failure modes that should be considered and their interrelationship is dependent on the availability of large data bases on failure modes in similar parts. These data bases are not adequate and in most instances do not exist for today's military engines.

Session III

The response of manufacturers to user concern over current engine reliability was addressed from a design standpoint in papers No. 8, 9 and 10 and increased testing in papers 13, 14 and 15.

Paper No. 8 by J.P.Marechal discusses the CFM-56 development program and the actions being taken to insure improvement in engine reliability and maintainability. Paper No. 9 by B.L.Koff discusses the four major requirements for engine design. Simply stated the first three, performance, weight and cost are easily determined early in an engine development program. The last, reliability, of which durability and maintainability are part, is not fully known until the engine is well into service. To maximize early reliability the design must be based on an accurate definition of mission requirements and accurate trades between competing performance, weight, cost and reliability requirements. Once the optimum configuration is chosen rigorous attention to detail is required to insure proper execution in the design process. Test and evaluation programs are directed at proving the design and correcting deficiencies. Cyclic endurance testing which tax major elements of the engine have been helpful in defining weaknesses. Component test and analysis are used to backup the engine endurance testing which cannot reproduce all expected conditions in actual flight in all parts of the engine. Advanced instrumentation and special test equipment are being used to determine the actual condition of parts under operation and to accomplish special tests such as foreign object ingestion. Tracking the first engines entering service and observing the condition of parts as usage time accumulates and factoring this experience back into both correction of current problems and new engine design requirements will lead to continuing improvements in reliability and maintainability for the future.

One of the approaches to achieving higher reliability has been to utilize redundancy. Paper No. 10 by J.C.Rennesson discusses this approach and the precautions which must be taken. The requirements for redundancy in single engine aircraft are more severe than in multiengine transports. Redundant systems present difficulties in isolating faults during maintenance and produce increases in cost.

As noted in B.L.Koff's Paper No. 9, endurance testing has shown to be beneficial in helping to uncover weaknesses in the design. Paper No. 14 by B.Devoge discussed the design and results of the endurance test program on the Olympus 593 afterburner and nozzle assemblies.

Paper No. 15 by B.J.McDonnell discusses the accelerated mission testing of the F100 engine for the F-15 and progress to date. The test program was divided into four parts: (1) determination of the actual mission profile, (2) definition of test conditions, (3) accomplishment of the tests and (4) correction of problems exposed. Perhaps most significant to the writer of this report was the number of full throttle excursions which actually occurred based on measurement. This usage showed that the military qualification testing requirements were significantly out of touch with usage. The testing has been successful in accomplishing the objectives sought and in conjunction with "lead-the-force" engines has been responsible for extension of the maximum overhaul operating time limits. These tests are not the panacea to achieve good reliability but they represent a significant step forward.

Paper No. 13 by D.Dini reports on basic test work on an engine to determine its tolerance to foreign object ingestions and the need to continue such test work to determine the effect of structural damage caused by transient loads. A procedure for undertaking such transient load tests by using a gas shock tube is discussed. The need for development of test facilities capable of simulating the icing environment of helicopters is stressed.

Session IV

Engine health monitoring and engine diagnostics have been in use by commercial airlines for a considerable period as discussed by P.Chétail in Paper No. 12. The growth and success of these techniques has led to their becoming

a fundamental part of commercial airlines engine maintenance programs. Their usefulness has come from observing the trends that had been developing prior to an actual failure and then applying these lessons to corrective action on others as soon as the same symptom(s) appear. Continued success and growth of such techniques will require even closer collaboration between users and engine manufacturers. The methods currently being utilized for engine health monitoring in the French Air Force are discussed by C.Sprung in Paper No. 3. The preliminary results of T-38 on board engine health monitoring equipment test evaluations are presented by K.E.Eickmann in Paper No. 18. The results have been gratifying and continued testing is planned. A prototype activity based on hand processed trend monitoring of engine performance on a group of military transport engines has proven immensely satisfying in reducing secondary damage and maintenance cost.

Paper No. 16 discusses the results of experimental testing of an engine with known faults to determine the detectability through normally measured gas path performance parameters. Problems of fault detection and related sensor location are covered. The growing technology in electronic digital computers will bring about totally electronic fuel control systems. Such systems lend themselves to engine diagnostics and trend monitoring with minimum increases in complexity and cost. These possibilities are discussed in Paper No. 17 based on flight test experience with prototype V/STOL fighter aircraft.

5. CONCLUSIONS

The price of each new engine generation has increased with the user demands for improvements in engine performance. The performance advances have been achieved, apparently at the cost of increased complexity and reduced reliability. Today economic forces (increased fuel costs, higher operating and support costs and poorer reliability from newer engines) have produced the need for both users and manufacturers to re-evaluate their priorities on performance and engine reliability in context with long term objectives.

Comparing historical data of the major causes for engine unreliability for a variety of engines leads to the conclusion that the same types of problems are repeated in every engine type and generation. Comparison of users data on major causes for unscheduled removals for the same engine and aircraft combination show that certain users have problems not being experienced by others. These data suggest that variations in operational usage, environment, engine age or detailed maintenance practices must also be considered as contributing to engine unreliability.

Actual military engine usage differs substantially from that used as the basis for design. Recent studies, both reported during this conference and underway in the United States, show that major thermal cycles in actual engine operation are as much as 10 times higher than originally considered in the design process or controlling military specifications. Certainly, precise definition of the way engines are actually used is needed to properly address low cycle fatigue and stress rupture. Data of actual engine thermal cycles in each aircraft type on different missions are needed to update engine specifications and design criteria.

In discussing engine reliability the engine is frequently considered alone and not as a part of a propulsion system. If detailed historical studies had been accomplished, it is the writer's belief that subsystems elements such as starters, oil system components, ignitors, instrumentation, air bleed valves, etc., are often major sources of line maintenance problems, aborts, cancellations or delays, Growth in overall system reliability will be contingent on improvements in these areas also.

Comprehensive data feed back to manufacturers of engine and part failure information including part time/cycles, mode of failure and primary or secondary involvement is needed to improve engine reliability after entry into service. All too often, redesign action can be initiated on the basis of too little information. Comprehensive research into the causes for unreliability covering many years and many engines, particularly for the military services, will be needed to understand what changes in specifications, regulations, operating procedures, design practices and maintenance programs are required. Certainly the determination of the cost to the users of poor reliability is a first step.

ENGINE STRUCTURAL INTEGRITY PROGRAM (ENSIP)

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ABSTRACT

The Air Force is considering developing a new Military Standard for turbine engines. This standard is aimed at providing overall policy and requirements for turbine engine structural development during the entire systems life cycle. This paper outlines a first review of the tasks involved in the Standard. Specific items such as duty cycle and tests concerned with fatigue considerations are noted.

INTRODUCTION

The Air Force has for some time been concerned with achieving the maximum service life from its turbine engines. Several recent experiences have led to the recent need to relook our development and acquisition procedures. Because of the vital aspects of fatigue and structural design to this activity a special program to develop a Military Standard is being considered. This program would have the goal of outlining in a Standard (known as ENSIP) the key tasks and programs necessary to achieve satisfactory service life without unduly sacrificing the performance objectives of the engine and system.

CONCEPTUAL PHASE - TASK I

This is the embryonic phase of any new weapon system. It is the first stage in the long process that leads to a fully developed and operational system. During this phase, studies are made to determine what capabilities are required to achieve the desired operational needs.

The engine under consideration may be existing inventory engines, a derivative model of an existing engine or an advanced design.

. Because of the fluidity of the system design, we envision that engine structural design will not be heavily influenced in this phase by specific usage considerations but by generic usage considerations i.e., Fighter, Bomber or Transport usage.

The major structural considerations to be explored are the influence of engine weight on system performance and some preliminary definition of the engine duty cycle which is key to fatigue life design. These data will be obtained through study by the Air Force and system contractors and provided to engine contractors in preparation for the validation phase. This effort could be considered as Task I of ENSIP as shown in Fig 1.

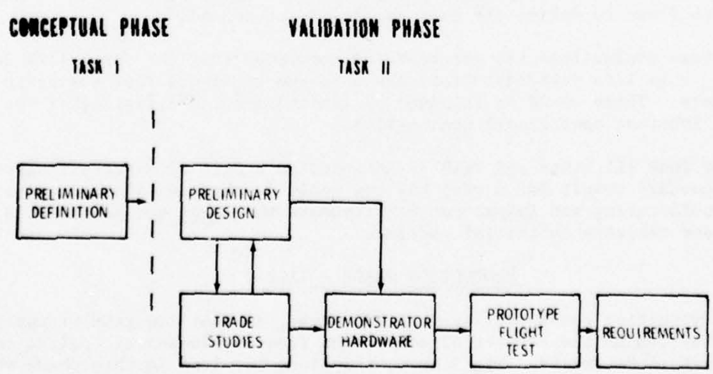

FIG 1. CONCEPTUAL AND VALIDATION PHASES

VALIDATION PHASE - TASK II

This phase sees the first system hardware being designed and built. If the program requires it, prototype aircraft may be constructed and evaluated as in the current Advanced Medium STOL Transport (AMST) flight test.

It is in this phase that the first concrete efforts regarding engine structural design result. In the preliminary design, the duty cycle defined in Task I will be used to design the life limited parts. Trade studies using the performance, life and weight will be made to further define what the true objectives of the engine should be. This effort results in demonstrator hardware as shown in Fig 1.

If significant technology advance is necessary to achieve the requirements, the demonstrator hardware may be prototype engines for ground and flight evaluation. Limited endurance testing will be a goal of having a safe flight test. The significant ground test activity will be aimed at defining the internal environment in which the parts must work as this information is critical to successful calculation of the structural life. Lesser technology advances will only require some structural rig test to new conditions.

The prototype flight testing, while primarily for system test, should also provide updated information on engine usage during the projected mission of the system.

The final output of this Task II activity could be defined structural requirements for proceeding into full scale development.

FULL SCALE DEVELOPMENT PHASE - TASKS III & IV

In this phase, the system is defined and work is proceeding to develop an operational capability. Two essentially parallel structural tasks are performed: structural analysis and structural tests. The process starts as shown in Fig 2 by review of the prototype system and update of its design. Engine trade studies are updated consistent with updated system requirements to insure that the best trades of performance, life and weight are being made.

FULL SCALE DEVELOPMENT PHASE

TASK III AND TASK IV

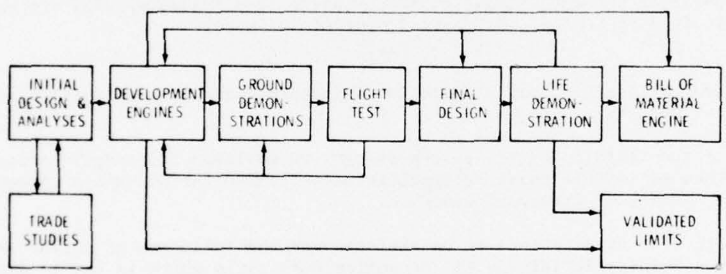

FIG 2. FULL SCALE DEVELOPMENT PHASE

Test hardware and engines are placed into ground test demonstration to aim at safe operation for the subsequent flight test program. These initial tests are limit type tests such as overspeed and overtemperature to ensure that no major design deficiencies exist. After flight test is underway, the projected mission may be flown to define the duty cycle for the engine.

It is only after these evaluations are successfully concluded that the first life demonstration testing be undertaken. Full life component tests could be run to ensure that the basic capability of the machinery is adequate. These would be followed by limited endurance testing of the engine to establish confidence in the inherent operational engine life.

The final result of Task III Tests and Task IV Analyses is a Bill of Material engine ready for low rate production. A secondary result but a very key one would be the preliminary definition of engine life limits and the manufacturing and inspection requirements that must not be violated if the engine is to be operated safely and reliably in initial service.

PRODUCTION PHASE - TASK V

In practice, the Production Phase overlaps the FSD phase, in that low rate production engines are produced to permit evaluation of the effects of conversion from developmental tooling to production tooling prior to high rate commitments. Fig 3 shows the steps involved in this phase which provide structural assessments for accelerating the engine production rate.

PRODUCTION PHASE

TASK V

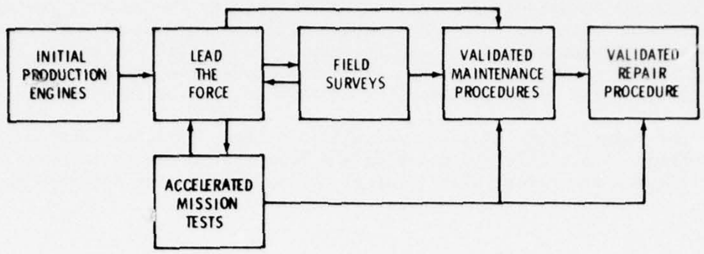

FIG 3. PRODUCTION PHASE

Initial production engines would be subjected to two distinctly different but complimentary structural assessments: Accelerated Mission Testing (AMT) and Lead-The-Force (LTF) testing.

AMT is special ground testing of the engine that simulates the cyclic fatigue and stress rupture loading that the engine receives in operational use. Engine flight operating time which does not contribute to these structural loading modes is eliminated from the AMT ground test time which may compress the required test hours by 50-75% to achieve equivalent structural damage. The AMT program is keyed to actual field surveys, as it proceeds, to correlate initial operational usage with assumed AMT test cycle. Successful AMT results are used to key the high rate production decision process.

The LTF program places the initial production engine into an operational environment which intentionally operates a few engines on a flight hour schedule which is significantly in excess of that achieved by the general operational engine population. These engines are otherwise operated and maintained as other operational engines. They are selectively returned to an overhaul facility for detailed teardown and inspection to identify operational problems which were previously unrecognized by the inherent limitations.

Both AMT and LTF programs are intended to accelerate our understanding of the probable structural maturity of the engine over that obtained by prior practices and to enhance our understanding of maintenance procedures and repair practices which must be instituted for safe, reliable and most economic consumption of the inherent engine structural life limits.

OPERATIONAL PHASE - TASK VI

The entire ENSIP process reaches completion in this phase. If the program has been followed, this task would be mainly one of keeping check of changing missions and uses as noted in Fig 4. These changes can result in engine modifications that need proof either through component or engine test. Also, overhaul and production engines and procedures are periodically checked through AMT. The monitoring of missions can be achieved by field surveys and/or integral engine or airframe recorders.

OPERATIONAL PHASE

TASK VI

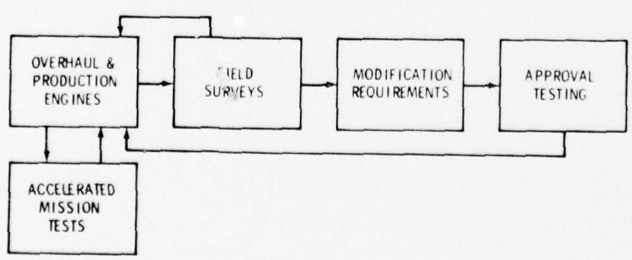

FIG 4. OPERATIONAL PHASE

SUMMARY

The continuing ENSIP development is primarily aimed at achieving a standard that will permit the Air Force to obtain engines with the best possible structural characteristics. Specific tasks for each phase of the development process may be identified and implemented. We intend to ask the assistance of industry and the other services in defining the tasks and determining the overall content of a standard. If we intend to maintain the outstanding record of the USAF turbine engine, we must extract the maximum life from the engine without compromising its basic structural capability and unduly impacting the performance level. Through the development of the processes and procedures of ENSIP and appropriate modifications to the engine general specifications, we believe we and the industry can achieve this goal.

This paper outlines one major thrust of the Aeronautical Systems Division towards improving the reliability of turbine engines i.e., life cycle design and test approaches to improved structural characteristics. Other thrusts are being taken in areas of improved engine controls and engine capability margins.

MILITARY ENGINE DETERIORATION IN SERVICE CONNECTED WITH LIFE CYCLE COSTS

by

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SUMMARY

Military engines community (manufacturers and users), in decades of experience, are well acquainted with the severity of mechanical deterioration and of the large amount of labor, skill and costs involved in maintaning engines in service at an adequate standard of efficiency and safety.

This paper attempts to afford some contribute to identify problems which are responsible for deterioration and to analyse structure of costs thus generated.

Main results, even only provisional nevertheless worthy of attention, are:

- Overall maintenance and repair life cycle costs appear to be comparable with the new engine cost.
- Performance deterioration seems not a problem of major concern for military operation, at least in the actual environmental context.
- Design must pay proper attention to "reliability" and "maintenability" concepts from the beginning and trade-offs should be performed in order to optimize the engine overall life cycle costs.

1. INTRODUCTION.

This report, with the exceptions of paragraphs 2.,3. and the Appendices, is based mainly on the data, ideas, concepts and findings collected and processed by the P.E.P. WORKING GROUP-08 on "AERO-ENGINE DETERIORATION IN AIR FORCE SERVICES" with the partecipation of experts from several nations and to which authors cooperated. Paragraph 2. and the Appendices are completely original. Paragraph 3. was thoroughly reworked in order to be exposed in the actual general form eventually usable by the reader.

Investigations were centered exclusively on jet-engines with or without afterburner, installed in combat aircraft. Data collection was the principal problem of the study due to difficulties of uncovering classified information and to the non-commonality of definitions, philosophies of maintenance and procedures of processing data which exist with each nation.

In spite of these adverse conditions the study was continued seeking to round-off asperities, adopting conservative hypotheses in order to save the effectiveness of data. Many topics are yet questionable but, it is believed, fundamental things are quite clear and, what is important, some are different from what was believed by common expectation.

Data were obtained for the engines illustrated in table I. (11 models, 6 families for a total park of about 4700 engines in exercise) ranking from about 3000 pound to 21.500 military static sea level thrust.

TABLE .I.

ENGINE MODEL	ENGINE FAMILY	TYPE OF ENGINE	A/B	THRUST T(lb) SLS NOMINAL	SFC(lb/lbfxhr) SLS NOMINAL	WEIGHT W(lb) / T/W RATIO(lbf/lb)	MANUFACT. LICENSEE	USERS DATA
1	"A"	TF BPR 1.5	NO	A/B --- MIL 21500	A/B --- MIL 0.60	2800 7.68	UK --	UK
2	"B"	TF BPR 0.7	YES	A/B 21300 MIL 12600	A/B 1.95 MIL 0.63	3600 350	UK --	UK
3		TF BPR 0.85	NO	A/B --- MIL 12000	A/B --- MIL 0.63	2700 4.14	UK --	UK
4	"C"	J	YES	A/B 17900 MIL 11900	A/B 1.96 MIL 0.84	3800 3.13	US IT	IT
5		J	YES	A/B 17900 MIL 11900	A/B 1.96 MIL 0.84	3800 3.13	US GRM	GRM US
6		J	YES	A/B 15800 MIL 10000	A/B 1.90 MIL 0.86	3500 2.86	US BL,GRM,IT,ND	BL,GRM IT,NL

TABLE I (continued)

ENGINE MODEL	ENGINE FAMILY	TYPE OF ENGINE	A/B	THRUST T(lb) SLS NOMINAL		SFC(lb/lbfxhr) SLS NOMINAL		WEIGHT W(lb) $\frac{T}{W}$RATIO(lbf/lb)	MANUFACT. LICENSEE	USERS DATA
7	"C"	J	YES	A/B MIL	15800 10000	A/B MIL	2.00 0.86	3400 2.94	US CAN	CAN
8	"D"	J	YES	A/B MIL	13700 9400	A/B MIL	2.03 1.01	3100 3.03	FR BL	BL FR
9	"E"	J	NO	A/B MIL	--- 5000	A/B MIL	--- 1.10	800 6.25	UK IT	IT
10		J	NO	A/B MIL	--- 4500	A/B MIL	--- 1.08	900 5.00	UK --	UK
11	"F"	J	YES	A/B MIL	4300 2900	A/B MIL	2.18 0.92	600 4.83	US CAN	CAN NL,US

2. ENGINE PERFORMANCE DETERIORATION.

Data obtained are of several types: precise statystical samples, single engine and average data. Therefore it was necessary to adapt the analytical statistics to these non homogeneous data. Some data are referring to a "deterioration concept", others to a "performance evolution concept"; some are more complete in the sense that from them is easy to deduce either one or the other concept. Anyway the processing was performed with very conservative hypotheses. Few data were found which was necessary to reject for various but clear reasons.

Performance evolution in military and maximum thrust and specific fuel consumption versus operation time is illustrated in fig. 1,2,3 and 4. These figures are related to engines of the family "C" (approx. of the same thrust and weight), which is a very typical engine family for combat A/C in use with several nations.

The visual examination of fig. 1,2,3,4 allows to formulate the following provisional judgements:

- Engine performance deterioration with time of operation (mil. and max. thust, mil and max S.F.C.) is clearly recognized.
- Military thrust (basic important parameter wich warrants the satisfaction of the military missions), during the operation time interval which was possible to investigate, is never below the nominal value.
- The dispersion of data, in terms of 3σ (σ = standard deviation), at least for the engines of the samples examinated is quite high, even with new engines. This fact is indeed well in line with the conclusions of ref. 5 (par. 5, FAN, COMPRESSOR AND TURBINE PERFORMANCE PREDICTION AS AN UNRESOLVED PROBLEM).

The following deductions descend immediately:

- The deterioration of the S.F.C. for "full power" has an impact on deterioration direct costs (total fuel consumption increase).
- The military engines seem to be designed and developed with a necessary "oversizing concept" on thurst, wich naturally is paid by the customer (user), being the price of the new engine, in general, proportional to static thrust. This phenomenon is a contribute to the indirect costs of the engine, caused by the thrust deterioration.

In order to establish more rigorously the dependance of military thrust deteroration with time of operation, different correlations formulas linear and non-linear were sought, taking the determination coefficient(see after) as a parameter of evidence of the eventual existence of this correlation. The best formula found is linear:

$$\left(\frac{\Delta T}{T}\right)_{ev.} = 5.424 - 0.0015 \times t \qquad 2.1.$$

$$r^2 = 0.615 \; ; \; r = 0.784$$

where:

$\left(\frac{\Delta T}{T}\right)_{ev.}$ = Deterioration of military thrust, in terms of percentage static, test cell, effective military thrust.

t = time, in operation hours.

r^2 = coefficient of determination. It is to remember that the "determination coefficient" is the ratio of the "explained" ("total" minus "residual") to the "total variation". That is the relative quantity of dispersion of statical data which is eliminated by the correlatior law.

r = coefficient of correlation.

The intercept of this right with the ordinata axis is the average "oversizing" percent of thrust for the new engine. The intercept with the abscissa axis gives the maximun "full mission age" of the engines. The term 0.0010xt is the expressions, in

function of time, of the deterioration.

Other correlations where obtained, in linear form, for maximum thrust, military and maximum S.F.C. evolution as follows:

$$\left(\frac{\Delta T}{T_{max}}\right)_{ev.} = 2.753 - 0.0012 \times t \qquad 2.2.$$

$$r^2 = 0.449 \; ; r = 0.670$$

$$\left(\frac{\Delta SFC}{SFC}\right)_{ev.} = 9.446 - 0.0052 \times t \qquad 2.3.$$

$$r^2 = 0.729 \; ; r = 0.854$$

$$\left(\frac{\Delta SFC}{SFC_{max}}\right)_{ev.} = 1.670 - 0.0022 \times t \qquad 2.4.$$

$$r^2 = 0.776 \; ; r = 0.881$$

where:

$\left(\frac{\Delta T}{T_{max.}}\right)_{ev.}$ = Maximum (A/B) thrust evolu tion, in terms of percentage static, test cells, effective maximum thrust.

$\left(\frac{\Delta SFC}{SFC}\right)_{ev.}$ = Military S.F.C. evolution, percent.

$\left(\frac{\Delta SFC}{SFC_{max}}\right)_{ev}$ = Maximum (A/B) S.F.C. evolution, percent.

Other data,especially for military thrust, were available for all the engines of table I. In fig. 6 is illustrated the envelope of such data. Many attemps (mono-, two- and three- dimensional) were therefore performed in order to find a general correlation formula between the military thrust deterioration and the operation time. All these attempt were unfortunate because the data are few and not sufficient to overcome the masking effect of the strong dependance of the deterioration with the military thrust (T_{mil}) and the engine weight (W).

The best found is the following expression:

$$\frac{\Delta T}{T_{MIL}} = \frac{37.28932 \times T_{MIL}^{0.66141}}{W^{1.14259}} \qquad 2.5.$$

$$r^2 = 0.755 \; ; r = 0.869$$

where:

T_{mil} = Nominal military thrust in lbs.

$\frac{\Delta T}{T_{MIL}}$ = Military nominal thrust deterioration in %.

W = Weight of engine in lbs.

The discussion of the 2.5, which follows, can be questionable, being based on the assumption that the correlation found is sufficiently universal. In the absence of other more complete investigations it will be nevertheless reported. Expression (2.5) can be written:

$$\frac{\Delta T}{T_{MIL}} = 37.28932 \times \left(\frac{T_{MIL}}{W}\right)^{0.66141} \times \frac{1}{W^{0.48118}} \qquad 2.6.$$

which (error in exponent 0.48 minor of + 3.9%, error in exponent 0.66 minor of + 1%)is very close to:

$$\frac{\Delta T}{T_{MIL}} = a \times \left(\frac{T}{W}\right)^{2/3} \cdot \frac{1}{\sqrt{W}} \qquad 2.7.$$

Formula (2.6) can otherwise be written:

$$\frac{\Delta T}{T_{MIL}} = 37.28932 \times \left(\frac{T_{MIL}}{W}\right)^{1.14259} \cdot \frac{1}{T^{0.48118}} \qquad 2.8.$$

which is very close to (errors in exponents of same magnitude as before):

$$\frac{\Delta T}{T_{MIL}} = a \times \left(\frac{T}{W}\right)^{7/6} \cdot \frac{1}{\sqrt{T}} \qquad 2.9.$$

Examinating qualitatively the expressions (2.7.) and 2.9.) one sees that deterioration increases with the ratio of thrust to weight and decreases respectively with the weight and with the thrust. This sentence appears perfectly reasonable.

Indeed as "bigger" is an engine (higher T and higher W), in a defined technological context ($\frac{T}{W}$ constant, see Appendix II), the expected deterioration is minor because the "scale" of the deteriorating factors (see par. 5) can be assumed constant.

Moreover formulas of the type of 2.7. and 2.9. express the skill of the manufacturer to build with given weight and thrust in function of the deterioration.

Same formulas are usable by the customer (user) for comparison reasons.

It is clear that the performance deterioration parameter ($\frac{\Delta T}{T}$) can be varied and adapted to the actual case, the same can be done with the performance (T) and constructive (W) parameter; it is not difficult with this line of throught to extend the result of this particular problem to the most important and basic problem of the industry and commerce of our era: "construct and purchase manufacts with predictable life".

What precedes is valid from a general point of view; there are other two particular important aspects:

- The deterioration rate at the beginning of the life of the engine.
- The "part power" deterioration.

These two points are covered by ref. 1 and 6. It will, for sake of brevity reported here an abstract from ref. 1. "Data concerning earlier versions of the "C" family engine suggest that a substantial portion of the deterioration that occurs with operational usage will occur in the first 50 to 150 hours of operation. This initial high rate will be followed by a lower rate of deterioration as operating time is accumulated. The conclusion of the analysis indicated that losses on the order of 1 to 2% in airflow and 2 to 4% in thrust could be expected as well as increases of SFC on the order of 1.5% at Military power setting. Higher losses in airflow and SFC could be expected at lower power settings. SFC levels of + 3.5 to 5% were reported from a limited sample of engines at 75% of normal power".

In conclusion from what precedes, it appears that, perphaps, the following expressions will describe the performance deterioration situation:

$$\frac{\Delta P}{P} = A \times t\left(1 - e^{-bt}\right) \times \left(\frac{T}{W}\right)^{n} \cdot \frac{1}{W^{m}} \qquad 2.10.$$

or

$$\frac{\Delta P}{P} = B \times t\left(1 - e^{-bt}\right) \times \left(\frac{T}{W}\right)^{q} \cdot \frac{1}{T^{p}} \qquad 2.11.$$

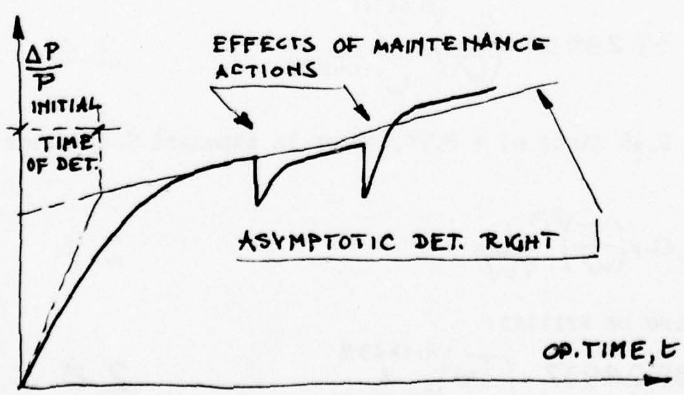
where:

P = general performance parameter.

$\frac{\Delta P}{P}$ = deterioration of P.

n,m,q,p,A,B,b = positive real numbers.

The graphic, deterioration time, will have the following appearance:



Another argument, for which it was possible to collect only few data, is how the overhaul or repair is improving the previous deteriorated performance. From fig. 5 it is clear that some positive improvement is statistically evidenced, but it is not excluded that even the overhaul or repair itself can, if not properly executed, provoke a deterioration. No correlations at all was possible to establish. The effects of all

the maintenance actions can be thought to be superimposed on the life-deterioration graph as historical events for each individual engine, giving an individual biography that, if obtained experimentally, could be rich of useful informations for the user and for the manufacturer.

3. AVERAGE LIFE CYCLE COSTS OF MAINTENANCE AND REPAIR VERSUS TIME, BROKEN DOWN INTO LABOR AND MATERIAL.

This analysis is focused on combat A/C engines (complete of accessories) with the aim to identify qualitative parameters which affect the direct maintenance and repair costs, trying to have an estimate of such costs using the data which were collected.

Data used are essentially those of the "C" family (see table I.) in service with Air Forces of different Nations, which are between each other comparable.

Financial data are actualized to the 1973-74 year.

Labor cost is in the area of 20 US$ per hour.

This study represents a preliminary effort and the related difficulties were principally those connected with different maintenance philosophies, doctrines and organisation concepts adopted by each Air Force. A more effective investigation, able to collect and process massive blocks of data, is believed necessary. Nevertheless, even if some data are only estimated, the "order of magnitude" conclusions are believed valid due to the very conservative hypotheses introduced in the course of the study.

3.1. AVERAGE LIFE CYCLE OF TURBOJET ENGINES INSTALLED IN COMBAT A/C.

The typical operational life E.L. of a turbojet engine for a combat A/C can be assumed to vary in the limits:

$$2000 \text{ op.hrs} \leqslant \text{E.L.} \leqslant 3500 \text{ op.hrs}$$

These figures are the results of an operational study valid under the following hypotheses:

- A/C are performing only peace time operation and remain in service for several lustra.
- No "attrition" is considered.
- All engines of the park (installed and spare engines) are sustaining a uniform operational load (due to the recycling flow of engines from overhaul and repair).
- The ratio s between spare to installed engines is varying from 0.3 to 0.5.
- The engines are of the "non modular" concept.

3.2. STRUCTURE OF DIRECT LIFE CYCLE COST.

Not taking into account pre-installation costs, each engine installed in an A/C can be thought to have the following direct life cycle cost structure:

$$\text{E.(NGINE)D.(IRECT)L.(IFE)C.(YCLE)C.(OST)} = \text{N.(EW)E.(NGINE)C.(OST)} \times (1+s) + \text{F.(UEL)O.(IL)C.(ONSUMPTION)C.(OST)} + \text{E.(NGINE)D.(ETERIORATION)C.(OST)} \qquad 3.2.1.$$

where:

s = Spare engine ratio (typically: 0.3 for twin, non modular, engines A/C; 0.5 for single, non modular, engine A/C; 0.2 for modular engines A/C).

Moreover, for the acquisition costs, we have:

$$\text{N.E.C.} = a \times T_{MILN}$$

where:

a = Specific cost of acquisition, customarily expressed in US$ per pound of nominal military thrust; depending from engine type, size, procurement year, quantity per lot, technology level, etc... (Typical average value for financial year 1973-74 about 50 US$ per pound).

T_{MILN} = Test cell, static, sea level, I.S.A., nominal military thrust (dry) in pounds

Consumption costs can be expressed as:

$$\text{F.O.C.C.} = (f \times \text{L.A.F.C.} + \phi \times \text{L.A.O.C.}) \times \frac{\text{N.E.C.}}{a} \times \text{E.L.} \qquad 3.2.3.$$

where:

L.A.F.C. = Life average S.F.C.

f = Cost of fuel per pound (typical for '73-74 about 0.10 $/lb).

L.A.O.C. = Life average specific oil consumption (typ. 2x10 lb/lbf hr).

ϕ = Cost of oil per pound (about 1$/lb).

E.L. = Engine life equal to A/C life divided by (1+s).

Deterioration costs are structured as follows:

$$\text{E.D.C.} = \frac{\text{E.L.} \cdot (O/Ml + O/Mm)\,\text{N.E.C.}}{\text{T.B.O.}} + m_a(M_a l + M_a m) \times \text{N.E.C.} \times \text{E.L.} + p(Ml + Mm) \times \text{N.E.C.} \times \text{E.L.} \qquad 3.2.4.$$

where:
T.B.O. = Average time between overhauls during a life cycle.
O/Hl = Average overhaul labor cost, measured in new engine costs, actualized.
O/Hm = Average overhaul material cost, in N.E.C., actualized.
m_R = Major repair rate, per op. hour.
$M_R l$ = Average major repair labor cost in N.E.C., actualized.
$M_R m$ = Average major repair material cost in N.E.C., actualized.
p = Average maintenance action rate, per op.hour.
Ml = Average maintenance action labor cost in N.E.C. actualized.
Mm = Average maintenance action material cost in N.E.C. actualized.

From expression 3.2.2. we deduce immediately that the engine direct life cycle costs are proportional to the nominal military thrust so that, known the thrust to weight ratio (see Appendix II.), it is easy to determine the costs per unit weight of engine which is a parameter more adequate for evaluation purposes.

The three cost components: acquisition cost, consumptions cost and deterioration cost, as we shall see after, are of the same order of magnitude and the second two are directly time dependent (proportional).

3.3. EXAMPLE OF COSTS SHARING OF ACTUAL TYPICAL MILITARY ENGINES FOR COMBAT A/C.

An appliance of the preceding chapter was performed in the course of the activity of the W.G.-08 having obtained conservative data for the engines of family "C". Analysis is developed on the basic maintenance scheme of fig. 7, which is giving a synthetic picture of the severity of effort for keeping an engine in operation. In this fig. are indicated the values of the labor necessary to each type of intervention. In fig. 8 is shown very synthetically the consumption of material pertaining to each maintenance action.

Answering to the question, as much universal these data are, it is here to remember:

- Some maintenance actions, not considered in fig. 7, were neglected: engine removals due to a non "phasing" between engine and A/C maintenance (av. 28% of the total number of removals considered); it was assumed that all modifications work was performed during an engine stop (which is not thrue, because many modifications are introduced expressely); calendar and special inspections (performed in the opportunity of special flight events as muddy landing, flight through hail storm, SOAP inspections etc...).
- The engines populations, from which basical labor and most of material data for the maintenance actions considered in fig. 7 were obtained, were relatively young (ab. 50% with an average of 270 hrs, ab. 50% with av.age of 1075 hrs) with an effective T.B.O. of less than 350 hrs.
- Overhaul and major repair spare parts incidence were determined, by the W.G. -08, in general for the engines in table I. respectively in the range of 0,11 + 0,15 and 0,02 + 0,04 of the new engine cost. Here are kept the lower limits.

Table II. summarizes the results of overall deterioration cost for two lives of 2000 and 3500 op. hours, for engines of family "C" (10.000 lbf thrust class).

TABLE II.

DETERIORATION COSTS IN N.E.C.

	LIFE OF 2000 op.hrs			LIFE OF 3500 op.hrs		
	LABOR	MATERIAL	TOTAL	LABOR	MATERIAL	TOTAL
OVERHAUL	0.1836	0.2200		0.2754	0.3300	
MAJOR REP.	0.0268	0.0370		0.0469	0.0648	
MAINTENANCE	0.1863	0.0850		0.3157	0.1488	
TOTAL	0.3967	0.3420	0.7387	0.6380	0.5436	1.1816
NEW ENGINE COST	0.5000	0.5000	1.0000	0.5000	0.5000	1.0000

Clearly, from table II., it can be sentenced that direct deterioration costs are in the order of magnitude of the new engine cost. From these data, easy calculations using the orientative data reported, permit to establish the hourly deterioration cost in the area of about 200 US$/hr.

In fig. 9 are illustrated the direct life cycle costs in function of time, with fuel at normal cost. Fig. 10 is illustrating same costs, with actual fuel cost.

3.4. OTHER EFFECTS ON LIFE CYCLE COSTS.

Very briefly it can be stated that other effects can be accounted on deterioration costs:

- A/C "attrition", expecially during war-time, lowers the engine life cycle: this fact is worthy of investigation reconsidering, perhaps, the necessity of manufacturing engines with very low life cycle without overhaul at all.
- Twin engine formula increases the new engines costs per aircraft (S. Appendix II), so, proportionally, it is increased the deterioration cost at required constant thrust.
- "Modular concept" is reducing the spare engines ratio s, due to the higher average T.B.O. of modules and because each module is overhauled and/or repaired more speedly. It is even to be expected a minor load (labor and material) of work per unit equivalent maintenance action.
- Performance deterioration is affecting total costs indirectly for two principal reasons: major average fuel consumption; over-sizing of new engines.

4. MAJOR SYMPTOMS CAUSING PREMATURE ENGINE REMOVAL AND ACCESSORY REPLACEMENT.

4.1. INTRODUCTION.

This matter needs much more theorisation in order to establish exact terms of comparison between same engine type in operation with different users and between different engine type and users and, above all, in order to obtain a rational understanding of what are symptoms. One important problem in this respect is to establish exact symptom levels of acceptance or rejection. The problem is that of a correct diagnosis well in correlation with a right prognosis able to identify the primary causes of the fault investigated. These problems are in the present time far from an adequate solution, nevertheless in practical terms it was possible to formulate the following sentences. Numbers in brackets are indicating the minima and maxima, averaged, of the unscheduled engine removals per 1000 flying hours as a consequence of the symptoms revealed.

4.2. PREMATURELY REMOVED ENGINES.

Foreign object damage (F.O.D. min. 0.25, max. 1.54) is, in order of importance, at the first place. Fuel and oil leakages occur quite often (min. 0.00, max. 0.80). Substantial reasons for removal are cracks, breakage, wear and tear (min. 0.00, max. 0.80) for a wide variety of parts. Contamination of the oil system is also to be mentioned (min. 0.00, max. 0.04).

Vibration (min. 0.00, max. 0.12) and stalls (min. 0.00, max. 0.24) occur practically in every engine type and family, however the engine removal rate is as a rule not very high.

Corrosion appears to be more a local climatic influenced problem (min.0.00, max. 0.07).

Problems inherent to specific engines also appear. For example, with reference to table I., on engine types 2 and 3, are evidenced broken rotor blades fasteners, on engine types 4,5,6,7, overheating, burning and failures in oil and sump systems are verified. On engine type 8, some displacement and shaking on compressor parts occurs;on engine type 11 some removals are due to problems in the hot section.

4.3. PREMATURE ACCESSORY REPLACEMENT.

Accessory problems are inevitably engine related problems. The comparison between engines of different families is practically impossible. By "C" family engines, for ex., the Temperature Amplifier must most frequently be replaced, problems exist also with the transfer gearbox, the torch igniter as well with the control alternator.

The chief faults found on accessories are in the cathegory of leakages, functional disturbances, wear and tear.

4.4. CONCLUSIONS.

It is necessary to establish on objective bases clear definitions for symptoms, studying each type in detail. For ex.F.O.D., which is a symptoms reported by all users, must be more specifically examined and reported in order to establish if it is due to bird-strikes, runway debris ingestion, flying object or a/c parts ingestion. Leakage, another common fault, has to be studied and classified per types, primary causes.

5. MAJOR ITEMS REPLACED AT OVERHAUL .

In order to fix the ten major items which must be replaced or restored during overhaul, two elements were considered:

- The replacement rate during O/H.
- The cost of the replaced (restored) part(s).

These two factors were multiplied by each other obtaining so a figure establishing the rank of importance of the items.

Data pertaining to the engines in table I. were collected by each country. Qualitative examinationand comparison of those data revealed:

- For the same type of engine not all items but a part (from 25 to 50 %) are common to other nations.
- Some items of the same engine type are common for several countries but that type of item is not reported by other nations.
- Some major item is just reported in one country but not in the others.

Reasons for those differences between countries seem to be:

- The average age of a same type of engine is different from country to country and major items list changes with engine total time.
- Engine configuration standard is different from country to country.
- Missions and environmental conditions are not identical.
- Maintenance philosophies may differ with the organisation. (For ex. parts which are scrapped instead to be repaired due to high level of parts in stock).
- Lack of realistic statistics especially established to meet the exact definition of major item (material and labor cost time replacement rate).
- Different origin of manufacture for parts.

In spite of this dispersion of concepts we can assume tentatively that the following are the major items which are replaced at overhaul with the main reason for change.

a. Compressor stator vanes and rotor blades for corrosion, affecting engine performances.
b. Compressor casings in light alloy for corrosion and/or distorsion.
c. Compressor discs for fatigue cracks.
d. Light alloy engine front-frame for cracks and corrosion.
e. Combustion chambers for deterioration. (Burns in liners, some metal pieces can deteriorate heavily the turbine).
f. Turbine nozzles for cracks and distorsion.
g. Turbine blades, especially of the first stage, for corrosion, burns and cracks.
h. Light alloy gear boxes, especially transfer gearbox, for distorsion, cracks and leaks.
i. Afterburner section for burns. (Liners, flameholders, cold and hot flaps).
j. Spray-bars for distorsion and cracks.

These parts are mostly common to all types of engines examined and exist already since a long time, but no permanent fix has been set up to now, despite of cooperation between users and manufacturers.

6. DESIGN CONCEPTS AND RELATION TO PERFORMANCE DETERIORATION.

All engines currently in active service were initially designed to provides a substantial performance improvement over their predecessor. Improved performance was sought through increases in temperatures, pressure ratio, mass flow and reducing weight of component parts. Inherent within this process was effort to permit higher allowable stress levels at elevated temperature.

These advances inherently brought increase in engine complexity and cost particularly as efforts were made to optimize engines for multiple applications or multi-mission roles. It is perhaps to believe that increased part lives and reliability were all too often sacrificed, at least partially, in an effort to achieve the ambitious goals sought in terms of performance. In order to achieve high mass flows and efficiency advanced airfoils were developed which were generally thinner and which were more sensitive to the effects of erosion. Lightweight materials such as aluminum and magnesium are sensitive to corrosion even with coatings developed to prolong their life. Successful engine programs have been followed by even more advanced versions of engines of higher performance. In the majority of instances, higher turbine temperature was used in achieving the advanced ratings required for derivative aircraft.

Performance deterioration results from: a) wear in seal areas, b) distortion of aerodynamic shapes, (i.e. turbine nozzle bowing), changes in airfoil surface conditions (roughness and shape) as the result of corrosion or erosion, the restoration practices used for parts to prolong their service life, and the increased clearances or acceptable mechanical parts standards allowed in the overhaul and repair manuals over those set for production engines. Therefore, some level of performance deterioration will always be present in engines after usage as a matter of both policy and design. Current qualification specifications permit an acceptable level of performance deterioration after type test (150 hour qualification test).

Reasonable engineering judgement suggests that performance deterioration related problems will increase in the future from a combination of factors. The limited budget faced by the military services will probably produce the necessity for the reduction in operating and support costs. The extension of engine parts life through refurbishment and more limited depth of engine restoration are strong cost savings candidates. Advanced derivative engines currently in service indicate that the rate of replacement of hot section parts is very high and at an entirely unsatisfactory level. Even more technologically advanced engines may encounter the same problems with immaturity, high removal rates and high maintenance costs, until modifications can bring them to a reaso-

nable level of maturity. Continued monitoring will be required by all military services to insure that a) more definitive performance information is collected and analyzed, b) appropriate policy revisions are accomplished and c) that proper tradeoffs are made between initial performance and long term performance including maintenance and reliability costs.

7. IDENTIFY WHY COMPONENTS DO/DO NOT DETERIORATE FASTER.

Performance deterioration is not currently a problem of significant magnitude because of the following factors:

a) Mechanical problems are limiting the life of hot section parts to such an extent as to cause their replacement at relatively frequent intervals.
Example: Replacement of combustion chambers and first turbine nozzle guide vanes and blades completely at 200 to 400 hour intervals.
b) Significant compressor deterioration has been delayed by the necessity of replacing compressor stator assemblies due to corrosion and fatigue cracking.
c) New engines are oversized so that maximum performances, even trough deterioration, are maintained during the life cycle at an acceptable level.

If parts lives were longer the potential for greater performance deterioration would exist. There are some indications that erosion and corrosion of compressor blades is now reducing compressor flow capacity.

Each engine will develop a unique deterioration history. Careful study of both performance and parts consumption/refurbishment data is necessary to understand each engine's performance deterioration problem. The study of what is not done by maintenance organizations is of equal importance to what is done when attempting to determine the component(s) responsible for observed deterioration.

8. IDENTIFY FEATURES TO BE ENCOURAGED TO OVERCOME PERFORMANCE DETERIORATION.

The following features will help to reduce performance deterioration:

During Design

a) Avoid the use of materials in gas path parts that are sensitive to corrosion.
b) Avoid placing inlets close to the ground or in positions where they can collect erosive material thrown up by landing gear.
c) Design with realistic seal clearances that can be maintained in operation.
d) Consider coating for surface protection.

During Maintenance

a) Provide tooling to overhaul facilities in sufficient quantity and of high accuracy with which to measure critical clearances, flow areas, and engine performance.
b) Provide proper instruction concerning the rework of compressor airfoils and sufficient tooling to allow control of leading edge shape.
c) Provide instructions concerning chord loss restrictions and detailed tools for checking chord lengths.
d) Establish realistic part performance lives (as distinct from mechanical lives) and periodically review part usage to insure that parts are not being used beyond these lives.
e) Attempt to design more conservative airfoils particularly turbine which are less sensitive to adverse temperature patterns and are inherently more repairable.

9. PERFORMANCE AND CONDITION MONITORING

The objective of a well run engine maintenance program should be to keep the engine out of the shop/repair activity for as long as economically practical. Pre repair testing of runnable engines would be helpful in identifying performance associated problems that could be handled during the repair and prevent subsequent test stand rejection. Future engines should provide means for extensive non-destructive inspection such as borescope, radioisotope, eddy current, etc., and for component instrumentation during ground running. Hand recorded performance information would behelpful in monitoring performance changes and potential faults during operation but probably go beyond a reasonable level of work load for single pilot aircraft.

Recording equipment for engine and other aircraft subsystems performance monitoring would be helpful only as long as the equipment were accurate and trained personnel were available to monitor the results. This type of data would be most helpful in determining how engines are actually used.

10. CONCLUSIONS

"Engine deterioration in service" covers the aspect of maintenance related to the original design of the engine with the aim to achieve the best compromise in term of cost-effectiveness between cost of the maintenance, cost of acquisition and operative cost.

Maintenance produces performance restoration, reliability and life regeneration with some extent of costs.

The actual study tries to summarise in the following sentences the major common

aspect that were individuated with sufficient credibility.

1) The cost of fighter engine maintenance are dominated by foreign object damage, corrosion of parts and poor endurance (lives) of parts, and further by the cost for maintaining engine performances.
   The cost of engine maintenance over the operational life of the engine is approximately equal to the cost of acquisition.
2) It is anticipated that future improvements in materials will be used to obtain higher performance and not reduced maintenance cost or improve engine reliability unless the adverse impact of poor reliability and high maintenance cost are more fully explored and proper trade offs defined.
3) Many of the major problems exhibited in particular engines have exhibited themselves for long periods and through several models of the same engine type. This would appear to suggest less than full attention has been given to solving these problems by both users and manufacturers.
4) The engines studied exhibited modest level of performance deterioration which is not of major concern. The level of deterioration is controlled by the frequent replacement of hot section and compressor parts replacement togheter with an inherent oversizing of the engine which is paid with the acquisition cost of the engine.
5) There is a feeling that the pass off test performance checks do not adequately address significant aircraft operating power settings and that significant deterioration may go unnoticed as a result.

6) Some tradeoffs between retaining performance and need for lowering maintenance cost may need to be considered in the future, taking in account that the cost of acquisition, the cost of maintenance and the cost of fuel seem to concur with the same order of relevance in the total life cycle cost of the engine.

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ACKNOWLEDGEMENT.

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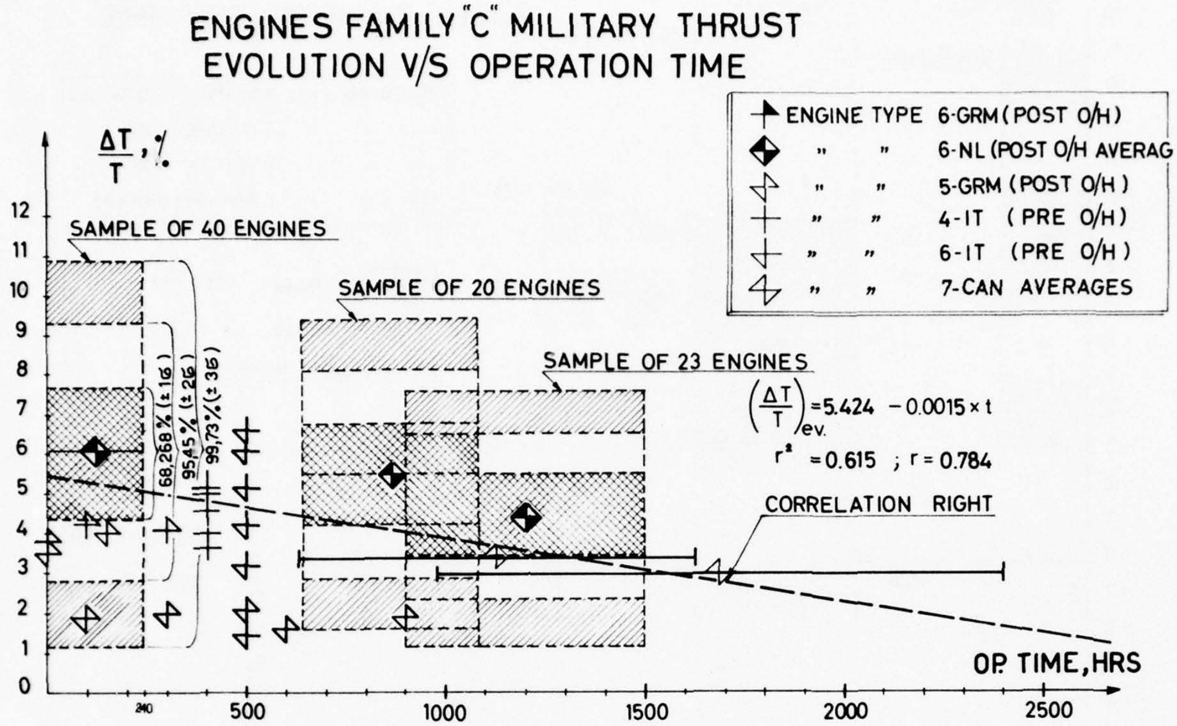
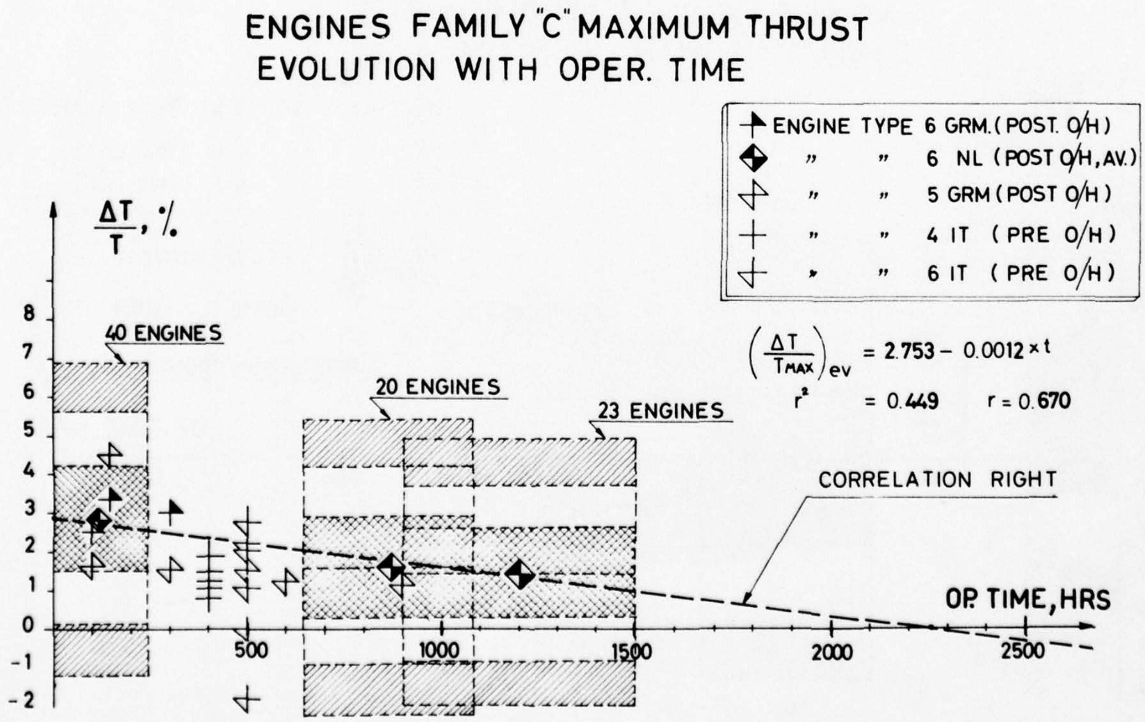


Figure 1



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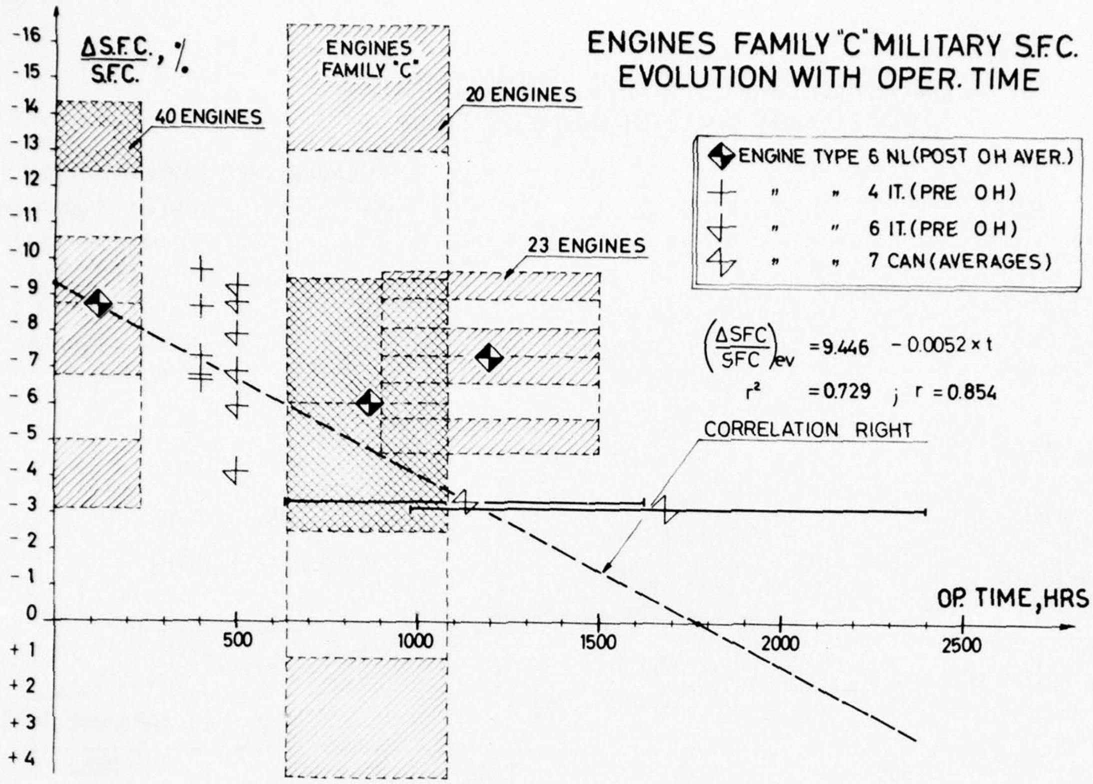

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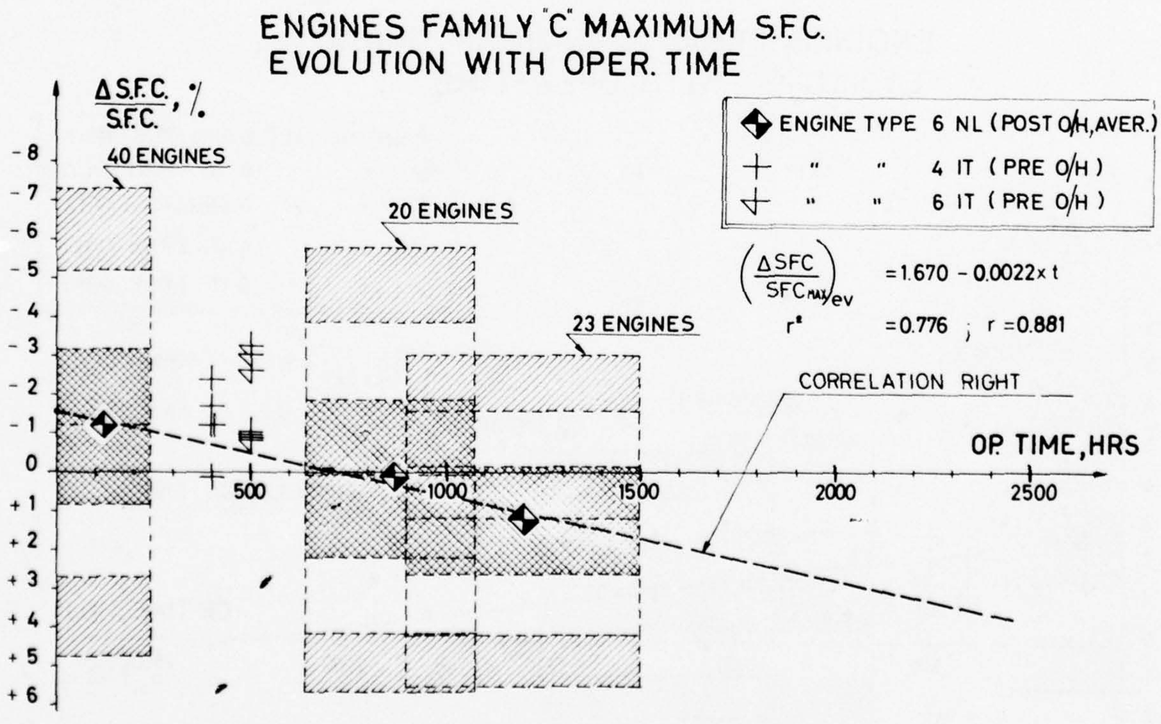

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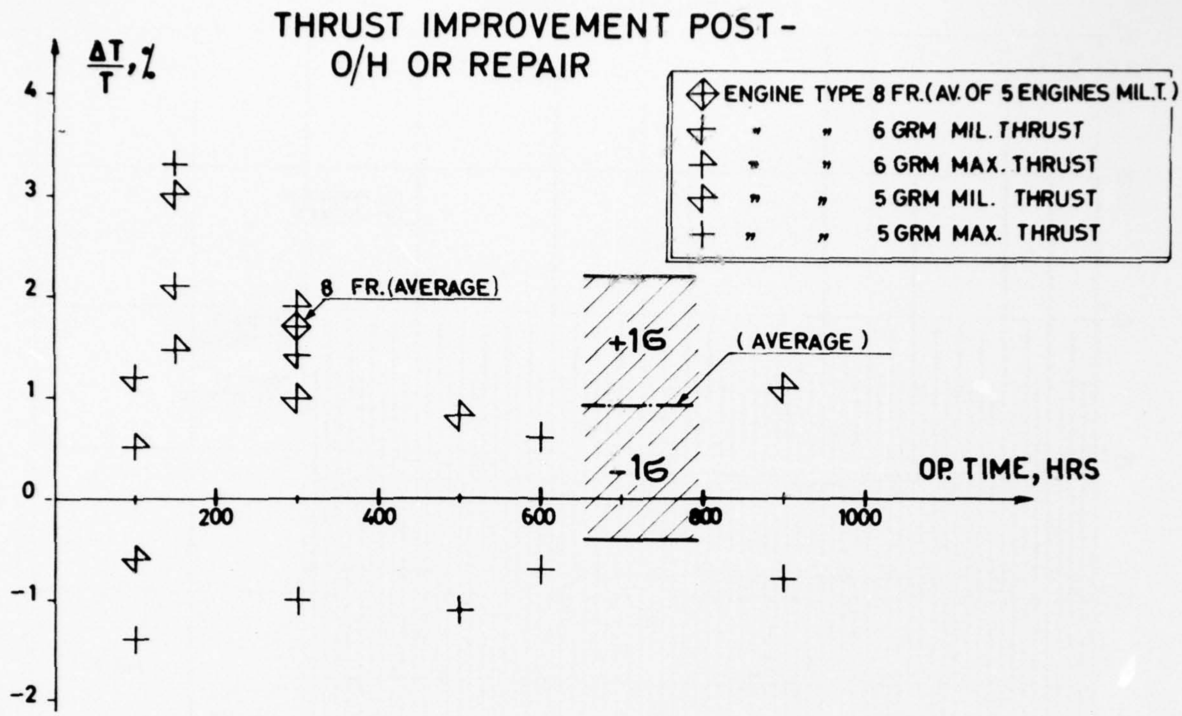

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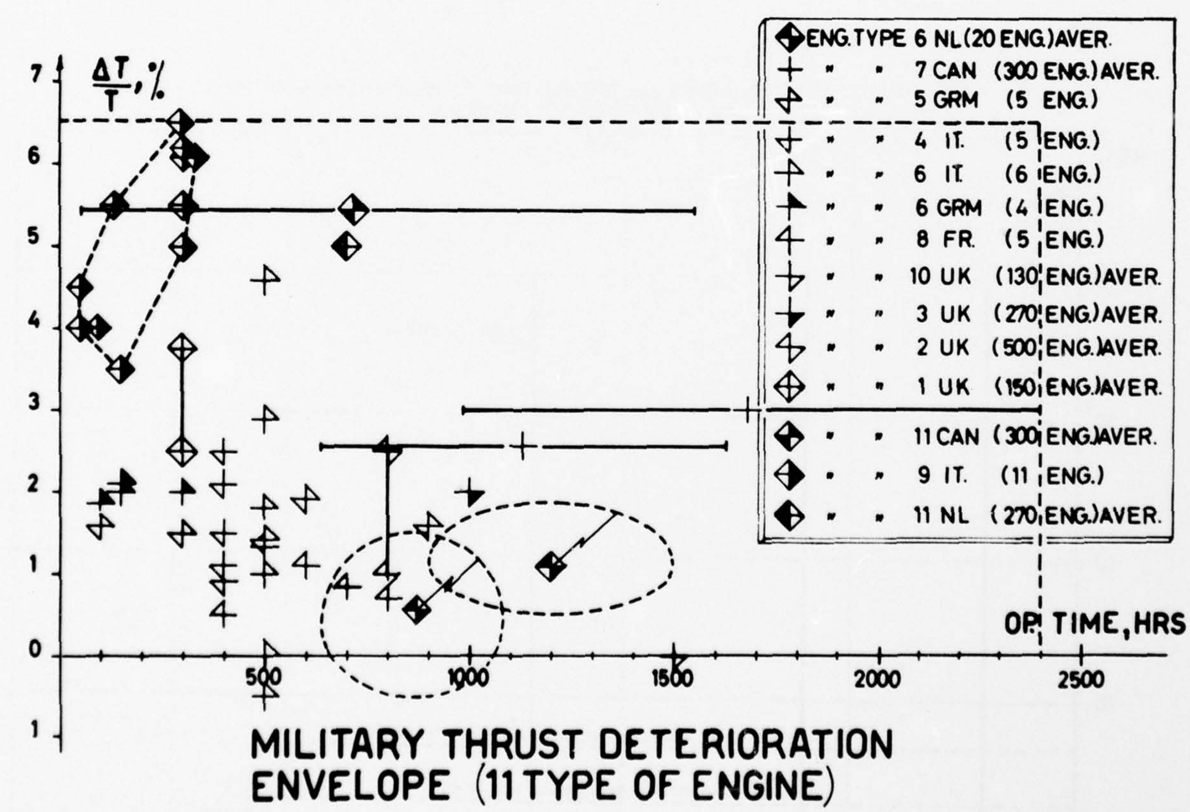

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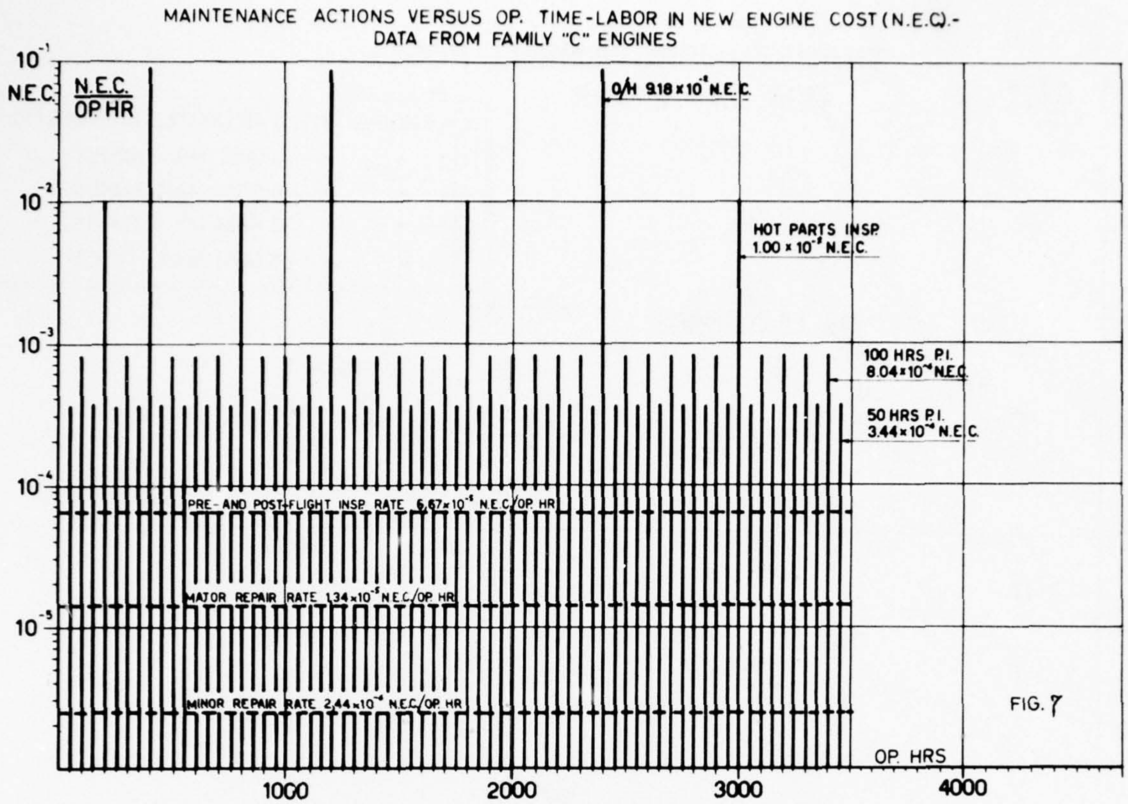

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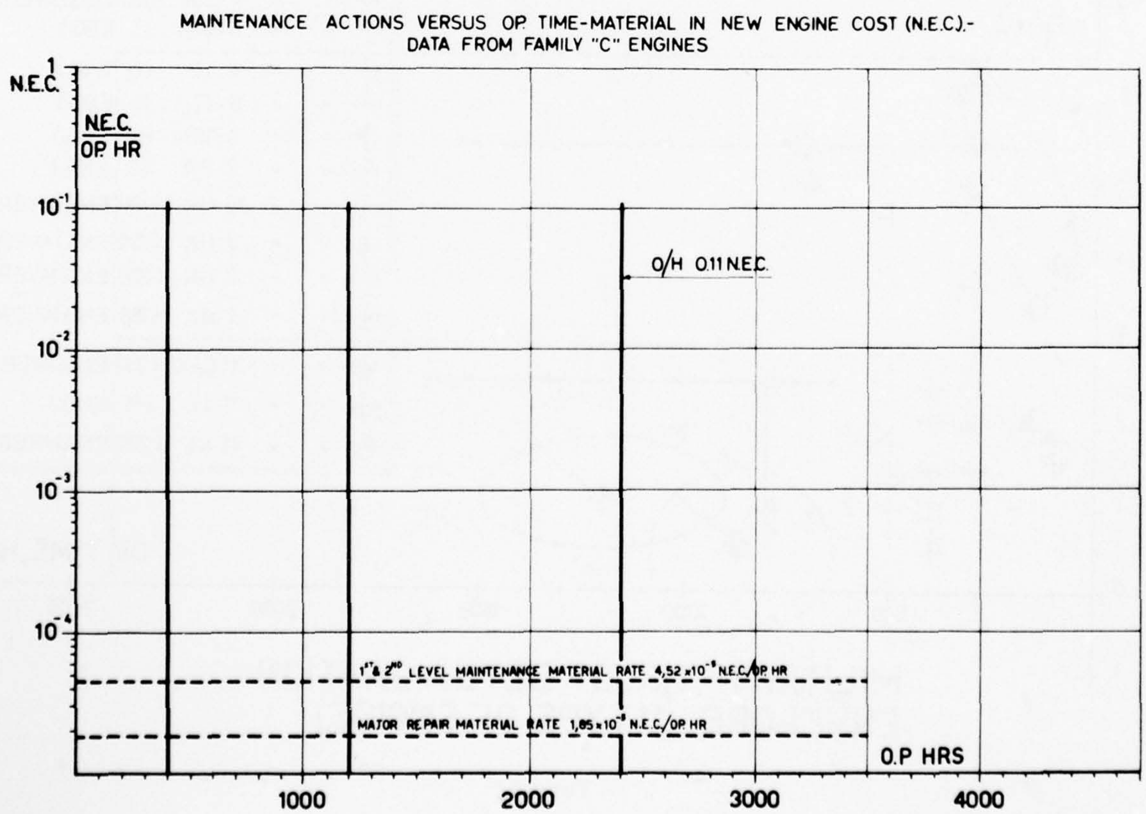
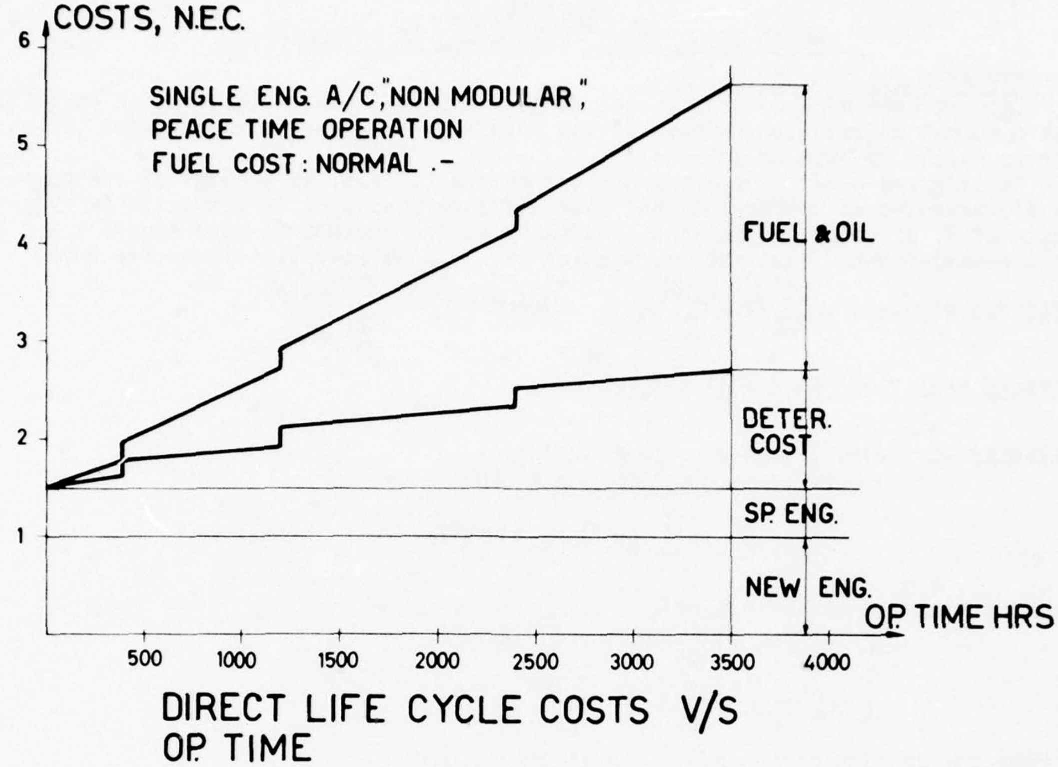


Figure 8



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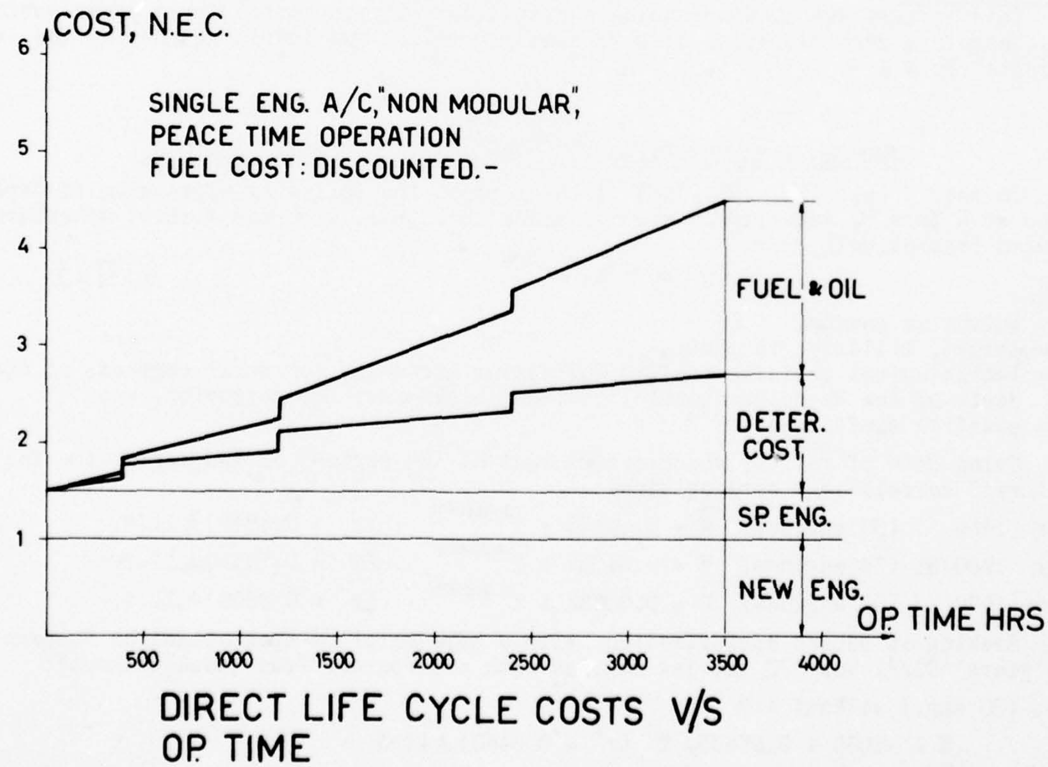

Figure 10

APPENDIX I.
NOTES ON CORRELATION FORMULAS OF PARAGRAPH 2.

a) FORMULAS 2.1., 2.2., 2.3., 2.4.-

Data at hand were not sufficient to give a significant determination coefficient with a normal correlation method. It was necessary to consider the "weight" of each generic couple of data as follows.

It is given a set of performance parameters P_i (f.e. as percent of the nominal value), measured at the operational time t_i, each couple P_i t_i being the average of a sample of N_i distinct individuals. Taking N_i as the "weight of each couple i and having pre-investigated that the correlation law is most probably linear, we have:

TOTAL VARIATION: $F = \sum_{i=1}^{n} (P_i - P_m)^2 N_i$; where $P_m = \dfrac{\sum_{i=1}^{n} P_i N_i}{\sum_{i=1}^{n} N_i}$

RESIDUAL VARIATION: $F_1 = \sum_{i=1}^{n} (P_i - a - b t_i)^2 N_i$

Following the usual "least squares method":

$$\frac{\partial F_1}{\partial a} = 0 = -2\sum_{i=1}^{n} (P_i - a - b t_i) N_i$$

$$\frac{\partial F_1}{\partial b} = 0 = -2\sum_{i=1}^{n} (P_i - a - b t_i) t_i N_i$$

Thus, solving the linear system:

$$\begin{cases} a\sum_{i=1}^{n} N_i + b\sum_{i=1}^{n} t_i N_i = \sum_{i=1}^{n} P_i N_i \\ a\sum_{i=1}^{n} N_i t_i + b\sum_{i=1}^{n} t_i^2 N_i = \sum_{i=1}^{n} P_i N_i t_i \end{cases}$$

we have the coefficients a and b, so we can calculate:

DETERMINATION COEFFICIENT $\quad r^2 = \dfrac{F - F_1}{F}$

CORRELATION COEFFICIENT $\quad r = \sqrt{\dfrac{F - F_1}{F}}$

b) FORMULA 2.5.-

This formula was obtained using a normal correlation method for a power surface fit. Therefore each triplet i of corresponding values was considered having the same "weinght" $N_i = 1$.-

APPENDIX II.
THRUST TO WEIGHT RATIO FOR "PURE JET" COMBAT A/C ENGINES.

In ref. 7 (pp. 132, 139, 140) it is proposed the following expression of dependance of W from T, referred to miscellaneous turbojets, with and without afterburner, deduced theoretically:

$$W = a + b \cdot T^{1,5} \qquad \text{A.II.1}$$

where:

W = weight in pounds.
T = thrust, military, in pounds.
a = technological positive coefficient taking accounts, for small engines, of the effects of low Reynolds numbers, minimum thicknesses for corrosion, etc.
b = positive coefficient.

Using data of ref. 3, which covers most of the engines of the world, the following "natural" correlations were obtained:

Year 1950 (30 engines) $W = 0.34465 \times T^{1.02680}$ $(r = 0.4849)$ A.II.2.

Year 1960/61 (78 engines) $W = 0.04352 \times T^{1.21021}$ $(r = 0.7912)$ A.II.3.

Year 1970 (52 engines) $W = 0.03759 \times T^{1.21589}$ $(r = 0.8509)$ A.II.4.

Seeking in second approximation, with a more detailed specialisation (separating for years''60/61 and '70 the jet engines with afterburner from those without):

1950 (30 eng.) without A/B

$$W = -1089 + 0.67435 \times T \quad (r^2 = 0.6480) \text{ A.II.5.}$$

1960/61 (22 eng.) with A/B

$$W = 173 + 0.32574 \times T \quad (r^2 = 0.7937) \text{ A.II.6.}$$

1960/61 (56 eng.) without A/B

$$W = -107 + 0.28062 \times T \quad (r^2 = 0.8652) \text{ A.II.7.}$$

1970 (19 eng.) with A/B

$$W = 748 + 0.23764 \times T \quad (r^2 = 0.9195) \text{ A.II.8.}$$

1970 (33 eng.) without A/B

$$W = -368 + 0.29900 \times T \quad (r^2 = 0.8906) \text{ A.II.9.}$$

Plotting the curves on graphs, it is easy to recognize the following fact:

- Curve II.1., which was proposed in the fifties, is well fitting values of engines of that age.
- In the fifties, there was no sufficient general manufacturing experience (low determination coefficient).
- In the sixties and seventies, experience is well improved (higher determination coefficients).
- Engines technology is improving with time in two respects: bigger engines manufactured and minor weight required for the same thrust.
- The practical law, weight versus thrust, is linear.

DISCUSSION

R.Smyth

Has the analysis of instrumentation data been used to monitor performance and deterioration between shop visits?

Author's Reply

The question of data assessment is the focal point of our work. We experienced great difficulties when analysing the data because of their lack of homogeneity. The point you are addressing might be answered in the future. Present day efforts are concerned with flexible operation demand and according to this work with operational availability of engines.

E.E.Covert

I would like to offer a comment, if I may, that would help perhaps bring some of these things into perspective.

There is a long time between the decision to make an engine and when the engine first goes into service. There is a long time after the engine is in service until one develops reliable statistics on its performance and reliability. Consequently from one view-point we are always looking on a moving target, because as we get better technology, we know better ways of handling the engine maintenance problem. And so sometimes these studies are dealing with engine maintenance procedures which evolved within a decade or more in the past.

So it is well to remember that there is a very large time constant associated with this.

P.Chetail

We have much appreciated the tentative formulation of thrust deterioration with operating time developed for straight jet engines. How would it apply to large by-pass ratio engines?

NOTE: Our experience seems to indicate that the deterioration rates for those engines (JT9D-7 or CF6-50) are larger than for earlier engine models (Dart, Avon, JT4, JT3D).

Author's Reply

Our statistics covered three turbofan (ref. Table I) with low by-pass ratio. For high dilution engines it is reasonable to think that correlation formulas should exhibit the same trends with proper deterioration parameters.

METHODES DE MAINTENANCE POUR AMELIORER LA FIABILITE DES PROPULSEURS

par

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RESUME -

1 - Point des études et méthodes de maintenance utilisées dans l'Armée de l'air.

2 - Etudes envisageables dès la conception du propulseur pour augmenter l'efficacité de ces méthodes et par là même augmenter la fiabilité des moteurs.

O - INTRODUCTION -

La part du budget consacrée à l'entretien des matériels ne cesse de croître au point de représenter un pourcentage très important de la valeur du parc de matériels opérationnels.

L' Armée de l'air se doit donc d'en améliorer la "rentabilité" par une augmentation de leur longévité et une réduction des charges de maintenance.

Le concept de "maintenance selon état" par opposition à celui de "maintenance préventive" a commencé à se préciser au cours des années 1965-1966. A cette époque, nous en étions encore au stade philosophique puisqu'il n'existait pratiquement aucun moyen technique ou si peu pour mettre en oeuvre les méthodes correspondantes.

La difficulté de cette évolution ne doit pas être sous-estimée.

Aujourd'hui encore, la sécurité des vols, souci permanent de tous les échelons de la hiérarchie n'a pas permis d'abandonner totalement le concept de "maintenance préventive" basé sur :

- des visites périodiques suivant un cycle bien déterminé,
- des limites de fonctionnement,
- des remises en état systématiques.

L'innovation consiste à supprimer ces visites périodiques et ces remises en état systématiques en assurant la sécurité par un contrôle plus ou moins continu du moteur en fonctionnement en vol ou au sol.

Si les moyens employés sont suffisamment efficaces, on économise ainsi toute une série de déposes et de démontages systématiques tout en assurant une sécurité des vols plus grande puisque la moindre anomalie est détectée à sa naissance et non à l'occasion d'une périodicité qui risque, dans un cas grave, de ne jamais être atteinte.

Nous possédons aujourd'hui une batterie de moyens dont certains sont généralisés et largement utilisés et d'autres où l'Armée de l'air est encore très timide, bien que le bien-fondé du moyen de maintenance en cause ait été démontré en laboratoire.

Les moyens les plus largement utilisés sont les suivants :

- Spectrométrie des huiles,
- Analyses des particules métalliques,
- Analyses des vibrations des réacteurs,
- Endoscopie,
- Ultrasons,
- Gammagraphie.

Je vais très rapidement passer en revue ces différents moyens, vous faire un bilan qui sera très superficiel compte tenu du temps dont je dispose, mais je suis prêt à répondre à vos questions à l'issue de l'exposé.

PREMIERE PARTIE

1 - SPECTROMETRIE DES HUILES -

1.1 - Politique d'emploi des spectromètres -

Les bases aériennes sont réparties sur tout le territoire. Des deux solutions extrêmes sont possibles : valait-il mieux avoir un seul laboratoire spécialisé ou au contraire plusieurs laboratoires installées à la limite sur chaque base ?

A) Solution "laboratoire central"

a) - Avantages

- meilleure utilisation du personnel spécialisé et du matériel,
- prix de revient du matériel moins élevé pour l'ensemble de l'Armée de l'air,
- qualité des résultats théoriquement meilleure.

b) - Inconvénients

- Délais beaucoups plus longs pour l'obtention des résultats dus surtout à l'acheminement des échantillons d'huile,
- servitudes engendrées par l'envoi des échantillons d'huile que ce soit par le service des postes ou par des moyens organiques de l'Armée de l'air.

B) Solution "laboratoire sur les bases aériennes"

a) - Avantages

-délais d'obtention résultats d'analyse très courts,
- l'échantillon peut être à la limite acheminé jusqu'au laboratoire, en cas d'urgence, par le mécanicien qui a fait le prélèvement,
- la capacité de mesures des appareils n'a plus besoin d'être très élevée. L'automatisation poussée, indispensable dans le cas précédent, n'est plus nécessaire. Les appareils sont plus simples donc moins chers.

b) - Inconvénients

- nécessité de mettre en place auprès de chaque laboratoire le personnel qualifié qui doit être instruit sur ce type de matériel,
- surveillance de la qualité des mesures est d'autant plus délicate qu'il y a plus d'appareils en service.

L'Armée de l'air a choisi la deuxième solution.

Il fallait par ailleurs choisir des appareils dont l'emploi est relativement simple : le choix s'est porté sur le spectromètre à émission.

La spectrométrie à absorption atomique a été abandonnée parce qu'elle réclame d'une part du personnel très spécialisé qu'il aurait fallu instruire et entretenir du point de vue qualification et d'autre part des laboratoires plus élaborés donc plus chers et dont la cadence d'analyses d'échantillons est trop faible.

1.2 - Résultats obtenus -

A titre expérimental, cette méthode a été utilisée à partir de 1968 sur des moteurs dont la technologie n'était pas du tout adaptée. Encore aujourd'hui, les moteurs surveillés par analyse spectrométrique des huiles sont loin de cette technologie idéale.

En particulier, le graissage à huile perdue des paliers entrainant une consommation d'huile importante a nécessité la mise au point d'un système analytique permettant la détection de très faibles teneurs en métaux et de trouver un paramètre représentatif de l'usure : la vitesse de pollution.

Ce paramètre est évidemment fonction de :

- la consommation horaire en huile,
- nombre de compléments de pleins d'huile,
- temps de fonctionnement du moteur entre deux prélèvements d'échantillon,
- la concentration en particules (p. p. m) de deux échantillons successifs.

Cette méthode a permis de supprimer de nombreux contrôles de la chaîne cinématique et est concrétisé par un guide de pannes, pour chaque type de moteur, introduit dans le manuel de maintenance.

Les résultats obtenus sur le dernier échantillonnage de matériels montrent que la méthode est encore perfectible.

Le pourcentage de matériels déposés sur anomalies constatées par A. S. H. (1), mais qui, au démontage, ne révèlent aucun défaut est encore trop élevé.

Par contre, aucun matériel envoyé chez le réparateur n'a présenté des défauts qui auraient pu être détectés par A. S. H.

Aussi, la sécurité des vols est bien assurée, mais l'Armée de l'air soit s'attacher à réduire si non supprimer ces retours injustifiés.

Les causes réelles de ces erreurs ne sont pas encore connues avec précision mais il est apparu récemment que l'huile neuve, contenait au départ des particules métalliques et cela est malheureusement toléré par la norme.

Deux actions sont possibles :

- tenter de supprimer ces particules parasites à la fabrication. Il apparait déjà difficile sinon impossible d'être aussi intolérant.
- il faudra donc s'assurer que les lots de fabrication soient homogènes et soient bien sur identifiés.

Il existe une autre cause, c'est la mesure de la consommation horaire d'huile.

Les études ont montré que la précision avec laquelle il était nécessaire de donner la consommation horaire était le dixième de litre par heure.

L'expérience montre que cette précision est très difficile à obtenir avec les moyens dont nous disposons, en piste, pour faire les pleins.

Nous recherchons toujours ce moyen précis, rustique et facilement utilisable dans le cadre de la mise en oeuvre d'un avion de combat.

La formule de calcul de la vitesse de pollution montre en effet que toute augmentation artificielle du facteur "q", toutes choses égales par ailleurs, augmente la valeur du résultat final.

(1) A. S. H. : Sigle signifiant analyse spectrométrique de l'huile.

Les mécaniciens ont une tendance à afficher une consommation supérieure à la consommation réelle. C'est ainsi que pourrait s'expliquer, aujourd'hui, les retours injustifiés de matériel.

Il serait évidemment très intéressant de comparer ces chiffres avec ceux obtenus par d'autres Armées de l'air ou Compagnies Civiles.

2 - ANALYSE DE PARTICULES METALLIQUES -

En complément de l'analyse d'huile, l'examen des dépots métalliques recueillis sur les filtres ou sur les bouchons magnétiques procure des indications sur le type d'avarie soupçonnée.

Certaines dégradations non perceptibles par l'A. S. H. ou d'autres méthodes, et liées à des phénomènes de fatigue le sont par cet examen.

Pour certains moteurs en service, tel le turbopropulseur TYNE, un catalogue très complet des différentes particules possibles existe.

Cette méthode exige du personnel très qualifié, et n'est pratiquement possible que lorsque la maintenance est très centralisée au niveau de l'Armée de l'air, l'idéal étant atteint dans le cas d'un seul atelier de révision pour le parc d'un type de moteur.

Il est possible bien sur de pallier cet inconvénient en expédiant les particules recueillies à un laboratoire dépendant de l'industriel, mais cela alourdit considérablement la procédure.

3 - SURVEILLANCE DES VIBRATIONS D'UN MOTEUR -

En matière de surveillance vibratoire, on distingue deux domaines très différents :

- le signal "hors tout" filtré, qui est une surveillance du niveau global de la vibration due essentiellement aux balourds des mobiles.
- l'analyse vibratoire qui permet de détecter certaines anomalies et avaries mécaniques à partir des composantes de signal "hors tout" non filtré.

3. 1 - Surveillance "hors tout" -

Le niveau du signal "hors tout" est représentatif de la tranquillité de marche du réacteur.

Cette surveillance n'est utilisée qu'à la sortie du réacteur de révision générale et permet de mesurer la composante vibratoire due à un balourd du mobile. Elle est abandonnée dans l'Armée de l'air au profit de la méthode suivante.

3. 2 - Surveillance par analyse vibratoire -

Cette surveillance qui est pratiquée au niveau des unités devrait permettre de déterminer l'état mécanique des organes internes du réacteur.

Pour ce faire, on utilise le signal vibratoire fourni par le capteur, mais au lieu de s'attacher seulement au niveau global de ce signal, comme dans la surveillance "hors tout", on le décompose et on analyse ses composantes.

Ceci n'est devenu possible que par l'introduction récente sur le marché industriel, de capteurs à cristal piézo-électrique à hautes performances.

L'analyse du spectre s'effectue sur les deux fonctions : vitesse efficace et accélération.

La décomposition du signal vitesse dans les basses fréquences autorise une surveillance étroite des paliers et éventuellement d'autres défauts. .

L'analyse du signal accélération vers les hautes fréquences permet de suivre les pièces tournantes délivrant peu d'énergie vibratoire mais possédant des vitesses angulaires élevées. Cela concerna plus précisemment :

- l'aubage du compresseur et de la turbine,
- la pignonnerie de la chaine cinématique,
- les accessoires.

Cette surveillance a été généralisée sur un parc de moteurs d'un même type (ATAR9C).

Si les résultats encourageants déjà obtenus se confirment, la méthode sera étendue à d'autres moteurs en service et surtout adaptée dès leur mise en service sur les moteurs nouveaux.

3. 3 - Point d'avancement -

A ce jour, la méthode est concrétisée pour le réacteur ATAR9C par l'existence d'un guide de signatures de pannes.

Ce guide repose sur l'exploitation d'un signal vibratoire issu du carter d'entrée et dont le traitement a été effectué :

- en vitesse de 0 à 500 Hz (unité : cm/S)
- en accélération de 0 à 20. 000 Hz (unité : g)

Il faut noter qu'au-delà de 5000 Hz les signaux n'ont qu'une valeur qualitative (le diagnostic est effectué sur la présence ou l'absence d'un ordre sans tenir compte de son intensité).

Les signaux doivent correspondre à des adresses cinématiques connues.

En effet, chaque mécanisme du réacteur engendre des vibrations qui sont caractérisées par leur fréquence.

- Un balourd du rotor engendrera une pulsation par tour du rotor, donc se manifestera sur la fréquence de la vitesse de rotation (N).
- Un engrenage ayant "n" dents et lié directement au rotor donnera "n" pulsations par tour de rotor et se manifestera par une fréquence n x N. On dit que cette vibration est sur l'ordre n.
- Au niveau des aubages, il y a de faibles intensités vibratoires correspondant aux variations progressives des quantités de mouvement dans les tuyères constituées par les aubes fixes et mobiles.

Si les triangles des vitesses sont corrects l'écoulement des filets fluides a lieu en harmonie avec le profil des tuyères sans choc aérodynamique important.

Par contre, toute anomalie dans le profil des aubes (impacts, ruptures...) entraine une modification des triangles des vitesses : il y a alors chocs sur les aubages sains suivants autant de fois que l'étage possède d'aubes.

Pour les vibrations d'un ordre n qui n'ont qu'une seule origine, les accélérations ont en général un niveau de 1,5 g.

Pour les vibrations d'un ordre donné qui peuvent avoir plusieurs origines (une roue mobile de compresseur qui a un nombre d'aubes égal au nombre de dents d'un engrenage de la chaîne cinématique) le niveau est sensiblement plus élevé : 3 à 4 g.

Ainsi, au guide de pannes, on voit, par exemple, que :

- tout défaut à un étage mobile sera détecté sur la grille fixe suivante et inversement.
- pour discriminer un défaut d'aubes de celui d'engrenage il faut faire appel à l'analyse de l'huile ou de particules.

4 - L'ENDOSCOPIE -

L'endoscopie, compte-tenu des progrès immenses réalisés ces dernières années, reprend de l'intérêt.

- les orifices nécessaires peuvent être très petits : quelques millimètres de diamètre.
- la luminosité des appareils est bien meilleure que celle connue il y a une dizaine d'années.
- les fibres optiques permettent la réalisation d'endoscopes souples.
- et enfin le matériel est simple et contrairement aux méthodes précédemment exposées, l'endoscopie ne nécessite pas de traitement.

J'ouvre une petite parenthèse : ce point est très important. Très souvent des méthodes de maintenance selon état échoue parce qu'elle demande une infrastructure pour le traitement importante, qui n'est pas mise en place parce que trop chère.

Sur le réacteur ATAR 9K50, qui vous est présenté sur ces diapositives, l'endoscopie a trouvé sa première utilisation globale : création de 2 orifices sur le carter compresseur et de 5 dans la chambre de combustion.

Par ailleurs la visualisation à l'intérieur d'une enceinte est devenue d'une telle netteté qu'il est possible de comparer un état par rapport à un autre : par exemple l'usure de cannelures internes avec un endoscope à réticule gradué.

Le principe est simple : un réticule gradué est incorporé dans l'optique d'un endoscope classique. La comparaison des largeurs de sommet de dents comptées en graduations entre les flancs travaillants (qui s'usent) et les flancs non travaillants (qui restent neufs) donne exactement le degré d'usure et permet une interprétation de l'usure (par exemple phénomène de rotulage résultant d'un défaut d'alignement).

Dans des cas bien précis, le manuel de maintenance du réacteur donne directement le nombre minimal de graduations admissibles sur la partie usée de la dent.

Bien que la méthode soit universelle, il ne faut pas perdre de vue que chaque réticule doit être adapté aux dimensions des cannelures.

Les nouvelles performances des endoscopes permettent également l'association d'une chaine de télévision.

L'intérêt est rapidement apparu dès qu'il s'agit du contrôle d'un nombre d'images très important.

Ainsi le contrôle du compresseur de l'ATAR 9K50 où l'opérateur doit examiner près de 250 aubes en quatre positions de pénétration de l'endoscope, soit 1000 images est fait avec moins de fatigue et plus de sureté.

Voici deux réalisations, pour l'instant à l'état prototype, qui seront utilisées dans l'Armée de l'air.

5 - METHODES DIVERSES POUR DETERMINER L'ETAT DES MOTEURS QUE L'ARMEE DE L'AIR UTILISE SUIVANT BESOIN -

5.1 - Contrôle par ultra-sons -

L'emploi des ondes ultrasonores permet la détection des discontinuités internes ou superficielles dans les solides.

La méthode utilise la réflexion d'une impulsion ultrasonore sur la discontinuité.

La réflexion est maximale sur la séparation solide - air : c'est la cas d'une crique.

Dans le cas d'une inclusion solide, le facteur de réflexion est fortement influencé par la nature de l'inclusion et du milieu dans lequel elle se trouve.

Bien adaptée, la méthode est souple et relativement facile à mettre en oeuvre.

Les principales applications de la méthode sur les moteurs dans l'Armée de l'air ont été les suivantes :

- Recherche des criques de fatigue sur les pales de R. M. 1.
- Recherche de criques sur injecteur ATAR 101.
- Recherche de criques dans les disques de compresseur sans dépose des aubages.

Cette recherche se pratique sur tous les réacteurs ATAR 9 ou 101.

5.2 - Gammagraphie -

La gammagraphie est un procédé de contrôle de type radiographique dont le rayonnement provient d'une source radioactive.

Ces sources sont en général des isotopes artificiels tel que le cobalt 60 ou l'iridium 192.

L'isotope émet dans toutes les directions un rayonnement "gamma" qui traverse la pièce à controler et impressionne un film photographique en donnant un contraste fonction des variations d'épaisseur, de la présence d'une crique, d'une zone déformée, manquante ou usée.

Le film développé, on obtient sur le négatif l'image des défauts recherchés.

La gammagraphie n'est pas systématiquement utilisée en maintenance.

Elle a eu une application sur grande échelle pour contrôler les mélangeurs de réacteur ATAR 9K50 et rechercher des criques sur les oufes de refroidissement.

Cette opération a permis d'acquérir une bonne expérience du contrôle gammagraphique en mettant en évidence ses avantages tels que rapidité, simplicité de réalisation et d'exploitation et ses contraintes telles que la durée de vie limitée de la source et les précautions à prendre lors des tirs.

Voilà la batterie de moyens de maintenance dont nous disposons. Quels sont les progrès envisageables et comment orienter les améliorations ?

On constate qu'aucune méthode est une panacée à elle seule, mais que l'emploi de l'ensemble de ces moyens pourrait peut-être permettre un suivi selon état sans démontage.

Etant donné la complexité croissante des réacteurs, les constructeurs doivent s'organiser pour penser "maintenabilité" du moteur, à savoir son aptitude à la maintenance.

Un effort certain a été fait ces dernières années, mais qui ne me parait pas suffisant du moins pour les moteurs militaires.

Il est évident qu'une collaboration étroite doit s'établir entre le constructeur et l'utilisateur afin que le matériel soit conçu et réalisé en tenant compte de son entretien futur pour éviter des durées d'immobilisation pour recherche de panne et remise en état incompatibles avec une utilisation opérationnelle et économique.

Deux grandes voies sont possibles :

- améliorer les moyens de maintenance,
- et, surtout, concevoir les réacteurs pour les rendre plus aptes à cette maintenance.

Que peut-on envisager dans chaque domaine ?

1 - SPECTROMETRIE DES HUILES -

Afin de mieux localiser une avarie éventuelle par analyse spectrométrique, une étude a été menée sur un réacteur ATAR 101 sur lequel ont été utilisés des "traceurs".

Elle a consisté à mettre en place divers revêtements au niveau des portées de roulements sur chaque ensemble afin de distinguer sans ambiguité l'origine d'une usure. Ont été essayés l'argentage, le cuivrage, le chromage et le cobaltage.

Le moteur test malheureusement ne s'est pas usé aussi bien en vol qu'au sol et de ce fait l'expérience n'a pu avoir aujourd'hui une conclusion formelle.

L'étude est cependant poursuivie et il est envisagé de mettre en place certains éléments traceurs au cours des opérations de réparation.

Cette étude, si elle aboutit, aurait des applications très intéressantes dans le cadre de la maintenance modulaire.

On éviterait ainsi de renvoyer un moteur complet en révision. Seul le module concerné le serait.

Dans ce même cadre, la SNECMA a déjà réalisé un autre moyen pour discriminer le module défaillant.

Les points de prélèvements d'huile pour analyse sont specifiques d'un module et sont regroupés à un endroit accessible, le réacteur étant avionné.

A plus long terme, on peut imaginer un moyen de rendre l'analyse spectrométrique plus sensible en augmentant localement, si possible au point de prélèvement de l'échantillon, la concentration en p. p. m.

2 - ANALYSE DES PARTICULES METALLIQUES -

Afin d'améliorer la captation des particules et de mieux en localiser la provenance, la SNECMA a déjà mis au point un "piège à particules". Ce dernier est constitué d'un barreau aimanté entouré partiellement d'une toile filtrante recueillant les particules non magnétiques.

Les résultats sont très encourageants mais ce système ne pourra être monté que sur les moteurs nouveaux.

Par ailleurs, il est déjà possible et cela est réalisé en laboratoire, d'analyser les alliages composant ces particules au moyen d'une cellule à luminescence cathodique, adaptable aux spectromètres.

3 - ANALYSE DU SPECTRE VIBRATOIRE DES MOTEURS -

L'ensemble des bruits émis par les divers organes du moteur se propage d'une façon hétérogène dans les carters.

Les bruits sont perçus dans divers points avec un amortissement variable.

Pour l'ATAR 9C, il a été démontré que le capteur placé sur le carter d'admission percevait tous les signaux d'une manière acceptable, mais cette manière de procéder n'est pas idéale.

Sur les moteurs nouveaux, les emplacements doivent être choisis en fonction de la richesse du signal émis par les organes à surveiller.

Plusieurs capteurs dans ces conditions seraient nécessaires. La méthode de surveillance y gagnerait en précision et facilité de traitement.

4 - ENDOSCOPIE -

Des progrès sont encore possible dans ce domaine également.

Sur les moteurs nouveaux, les niveaux de température devant turbine et par voie de conséquence sur toutes les "parties chaudes" sont plus élevés. Ce fait entraine la nécessité de disposer d'un moyen pour détecter les défauts très fins difficilement décelables à l'oeil nu.

Dans ce cas, seul l'examen par ressuage fluorescent sous lumière ultraviolette est suffisamment performant pour atteindre cet objectif.

Un dispositif inspiré de l'endoscopie peut être utilisé en deux phases : la première destinée à la préparation des pièces et l'injection des produits de ressuage, la seconde pour l'examen sous lumière ultraviolette.

5 - ULTRA-SONS -

La miniaturisation des capteurs devrait permettre à l'avenir d'exécuter les contrôles sans démontage en utilisant les orifices prévus pour l'endoscopie.

Il est possible de fabriquer aujourd'hui des capteurs dont les dimensions sont très faibles (3 à 4 mm de côté).

DEUXIEME PARTIE

MAIS COMMENT ET QUAND UTILISER CES MOYENS JUDICIEUSEMENT ?

Aujoud'hui, on décide d'utiliser tel ou tel moyen au moment où on découvre pour la 1re fois un défaut, en essayant de l'adapter le mieux possible à la technologie du moteur, ce qui n'est pas toujours facile.

Peut-on prévoir les pannes et avaries et définir dans le même temps un moyen de surveillance ?

Oui, en effectuant une analyse de la totalité de la maintenance en précisant qu'elle comprend :

- la mise en oeuvre de l'avion. En effet, parallèlement aux opérations de pleins (carburant, huile) des opérations de surveillance peuvent être effectuées.
- les opérations d'entretien périodiques ou de réparations faites au niveau des forces.
- les opérations de révision générale ou de grosses réparations effectuées chez les industriels.

Il est bien évident que toute modification sur une des phases de la maintenance a des répercussions sur les autres.

Quels sont les différents chapitres de cette analyse ?

1 - ANALYSE DE LA LISTE DES ORGANES ET DES FONCTIONS -

L'établissement de cette liste est indispensable pour débuter l'analyse de la maintenance.

Le but de cette liste est de dégager les organes, peut-être leurs composants, dont il faudra surveiller l'état.

La liste étant établie, il faut en faire trois paquets :

- les organes qui peuvent être surveillés et même changés pendant la mise en oeuvre.
- ceux qui peuvent être régénérés au niveau des ateliers dans les forces compte-tenu des moyens qui seront mis en place. La décision doit découler d'une étude purement économique.
- ceux qui ne peuvent être traités qu'au niveau industriel pour les mêmes raisons économiques du cas précédent.

En regard de chaque organe ou composant, figure sa fonction. Il est en effet indispensable de connaitre la fonction détériorée afin d'apprécier l'importance d'une panne et l'utilité qu'il y a à y parer. Inversement la connaissance des fonctions est indispensable à la recherche des pannes puisqu'une panne a pour conséquence la cessation d'une fonction.

2 - L'ETUDE DES PANNES -

2.1 - Panne élémentaire -

C'est la panne de l'organe ou d'un de ses composants. Elle conduit au niveau de la fonction correspondante à une panne globale.

En face de chaque panne élémentaire, on doit trouver :

- la signalisation utile au pilote ou au mécanicien,
- la méthode de détection ou mieux de la tendance à la panne,
- le moyen à employer pour cette détection,
- le moyen de la corriger. Il est important d'étudier dans ce cas l'accessibilité en fonction du taux de pannes.

2.2 - Panne globale -

Ce genre de panne est très important puisqu'elle touche aux fonctions prépondérantes du moteur et par voie de conséquence de l'avion.

La liste des pannes globales découlent de celle des pannes élémentaires.

De cette analyse doit découler :

- les pannes les plus graves, celles qu'il faut savoir prévoir.
- pour une panne donnée un classement des organes par ordre de responsabilité possible.
- un tableau de recherche de pannes à partir des symptômes globaux qui permet de descendre au niveau de l'organe ou d'un de ses composants.

3 - LES FONCTIONS CACHEES -

Ce sont celles dont la panne n'est pas signalée directement ou indirectement au pilote.

Deux types de fonctions cachées peuvent être mis en évidence :

a) Type "redondant"

Deux organes sont en parallèle et fonctionnent en permanence. Ils ont chacun une fonction cachée, si l'un assure la fonction globale alors que l'autre est en panne. On ne voit pas la panne de l'un si l'autre fonctionne.

b) Type "occasionnel"

Certains organes ont une utilisation occasionnelle. Ils assurent par exemple des fonctions de secours à la suite d'une panne d'un organe, ou bien leur utilisation n'est simplement pas continue.

Les fonctions de ces organes sont dites cachées, car si on ne fait aucun contrôle, leurs pannes n'apparaissent qu'au moment de leur emploi, ce qui est fort gênant, surtout pour les circuits de secours.

C'est le type même de fonction qu'il faudra couvrir d'une tache de maintenance programmée.

4 - SYNTHESE DES RESULTATS OBTENUS PAR L'ANALYSE OBJET DES TROIS PARAGRAPHES PRECEDENTS -

A la limite, seuls les organes ou composants ayant une fonction cachée doivent faire l'objet d'une opération de maintenance programmée.

Pour les autres organes on ne doit retenir que les opérations susceptibles de préserver ou de restaurer leurs caractéristiques.

Le choix de ces opérations est très important puisqu'il conditionne la durée et la fréquence des indisponibilités du matériel.

C'est l'expérience qui doit guider ce choix, puisqu'au moment d'en décider, la phase de développement reste peu significative.

Les compagnies aériennes civiles et les constructeurs concernés ont constitué un groupe de travail, le M.S.G. (Maintenance Steering Group) qui a élaboré une méthode de détermination de ce programme de maintenance.

Cette méthode publiée d'abord sous le sigle MSG1 est devenue le MSG2 en 1970. Elle pose une suite logique de questions dont vous trouverez le schéma résumé en annexe et que je vous présente ici.

Sans expliciter entièrement ce schéma, on voit que la question n° 2 appelle un certain nombre de moyens que nous ne possédons pas aujourd'hui.

Actuellement, les dispositifs décelant les tendances aux pannes sont en effet peu nombreux.

C'est la raison pour laquelle nous portons nos efforts pour le développement des méthodes telles que l'analyse spectrométrique des huiles, et l'analyse de vibrations.

Ce plan de travail fait ressortir également les pannes des organes qui nous oblige à adopter une maintenance préventive à base d'opérations programmées ou de limite de fonctionnement.

La fréquence de ces opérations ou les limites ne peuvent résulter que d'études statistiques qui ne tiennent pas compte évidemment de l'utilisation individuelle de chaque organe.

Cette mesure doit être considérée comme un pis aller, car elle entraine des pertes de potentiel loin d'être négligeables.

Donner, par exemple, une limite de fonctionnement en heures à une turbine est aujourd'hui la seule solution que nous ayons pour préserver la sécurité des vols. Il faudrait développer, et cela existe déjà mais à l'état embryonnaire, des moyens de mesurer d'aussi près que possible le vieillissement réel d'une turbine.

Je me suis étendu sur l'élaboration de la maintenance préventive car c'est elle qui grève le plus les coûts de la maintenance et entraine des gaspillages qui peuvent être réduits si non supprimés à court terme.

Ceci ne veut pas dire que la maintenance corrective doit être négligée au moment de la conception.

L'analyse précédente cherchait à prévenir et détecter les pannes.

Maintenant il s'agit de les corriger.

Il faut donc :

- faire un diagnostic, donc mettre à la disposition des utilisateurs des moyens de recherche qui peuvent être les mêmes que ceux déjà cités.
- accéder à l'organe défaillant, le déposer, le reposer et remettre en condition le moteur. L'accessibilité découle de la conception même du moteur.
- et enfin contrôler la bonne exécution de l'opération et vérifier l'ensemble des réglages.

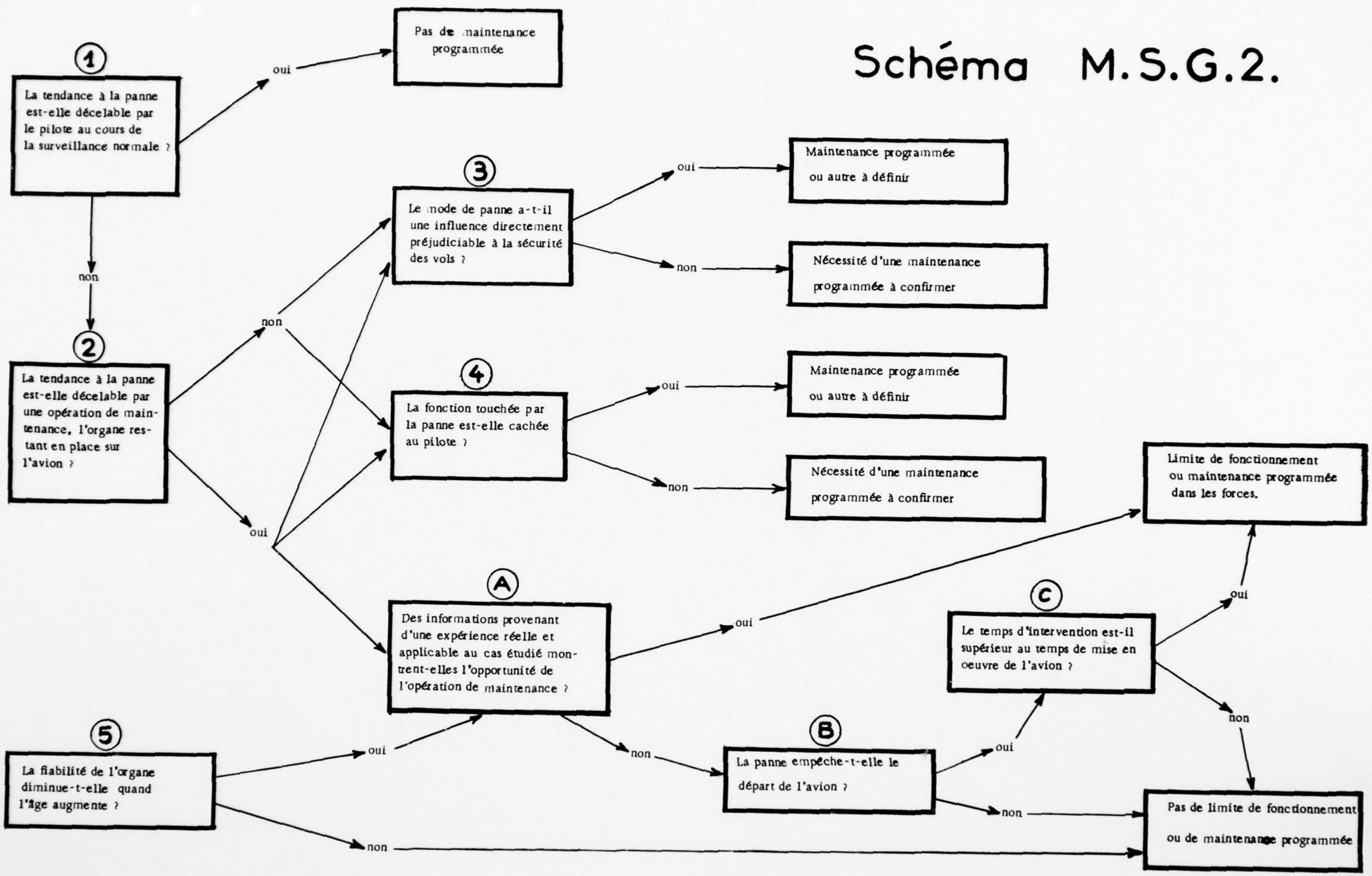
CONCLUSION

Je vous ai fait le point de ce dont nous disposions à ce jour et de ce que raisonnablement, on peut espérer avoir dans un proche avenir.

Mais pour les moteurs nouveaux le plus important est d'avoir un plan de travail pour prévoir la maintenance à la conception afin d'éviter les gaspillages en matériel et en main d'oeuvre que nous avons connus et que nous subissons encore.

Si les constructeurs avec les utilisateurs s'attachent à répondre aux questions posées par la méthode MSG2, la "maintenabilité" des réacteurs fera de grands pas.

Schéma M.S.G.2.


1
La tendance à la panne est-elle décelable par le pilote au cours de la surveillance normale ?
oui
Pas de maintenance programmée
non
2
La tendance à la panne est-elle décelable par une opération de maintenance, l'organe restant en place sur l'avion ?
non
oui
3
Le mode de panne a-t-il une influence directement préjudiciable à la sécurité des vols ?
oui
Maintenance programmée ou autre à définir
non
Nécessité d'une maintenance programmée à confirmer
4
La fonction touchée par la panne est-elle cachée au pilote ?
oui
Maintenance programmée ou autre à définir
non
Nécessité d'une maintenance programmée à confirmer
A
Des informations provenant d'une expérience réelle et applicable au cas étudié montrent-elles l'opportunité de l'opération de maintenance ?
oui
Limite de fonctionnement ou maintenance programmée dans les forces.
non
B
La panne empêche-t-elle le départ de l'avion ?
oui
non
C
Le temps d'intervention est-il supérieur au temps de mise en oeuvre de l'avion ?
oui
non
Pas de limite de fonctionnement ou de maintenance programmée
5
La fiabilité de l'organe diminue-t-elle quand l'âge augmente ?
oui
non

CIVIL AIRWORTHINESS REQUIREMENTS FOR POWERPLANT RELIABILITY

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SUMMARY

The airworthiness aim for a civil aero-engine is that under all likely flight conditions it will perform adequately or, if it fails, then the rate of failure and type of failure is not such as to hazard the safety of the aircraft.

It has been generally accepted that aircraft design should be aimed at a catastrophic failure rate for all airworthiness causes not exceeding a probability in the order of 10^{-7} per aircraft hour and from this it has been concluded that the airworthiness limit for major engine failures should be that no individual failure of an engine that could be potentially hazardous to an aeroplane should be predicted to occur more frequently than once in 10^{8} engine hours.

For lesser failures that only result in an engine shut-down, an in-flight shut-down rate in the order of 10^{-4} appears to be acceptable.

Service defects and in-flight shut-down rates must be monitored to ensure that an acceptable overall level of reliability is being maintained.

INTRODUCTION

Reliability means many things to many people. To the pilot, powerplant reliability means he expects to get the right power at the right time whenever he moves the power lever and the engine will then continue to function correctly throughout the flight, whilst the airline operator may be more interested in the economics of premature removals, flight delays, etc.

The Airworthiness Authorities must, of course, keep all these considerations in mind, but their over-riding consideration must always be related to the safety of the aircraft and its occupants. Thus our view of powerplant reliability can be summarised into three objectives.

1 Any failures of an engine that could hazard the aeroplane must be kept to an absolute minimum.

2 Loss of thrust in flight must not reduce the total thrust available to the aircraft to such an extent that the flight cannot be completed safely.

3 A normally operating engine must provide the thrust necessary for the aircraft to meet its scheduled performance and respond quickly and accurately to the demands of the pilot.

This Paper will consider each of these objectives and in the case of the first two attempt to indicate the levels of reliability that it is hoped will be achieved.

HAZARDOUS FAILURES

Over the years airworthiness targets have gradually been formulated that appear to be acceptable to the public on the one hand and can be achieved on the engineering side within the bounds of economic feasibility. Obviously the aim must always be to avoid catastrophic accidents altogether, but since absolute zero is unobtainable, it is more helpful when assessing the acceptability of a complex system to have a finite limit in mind and in the UK it has been generally accepted that aircraft design should be aimed at a catastrophic failure rate for all airworthiness causes not exceeding a probability in the order of 10^{-7} per aircraft hour or flight.

Various surveys of aircraft accidents over recent years indicate that broadly speaking engineering causes can be broken down into three major design areas, namely, structures (main structure and under-carriage), systems (hydraulics, brakes, instruments, controls, etc), and powerplant, and each area contributes roughly one-third to the accident rate. Therefore, strictly speaking, to arrive at our minimum level for powerplant reliability, we should divide the overall limit by three, but since there is always a degree of overlap between these areas, and in any case, it is unlikely that only the bare minimum will be achieved in each area on any one aircraft design, it is considered that the objective will be achieved provided the aim for each major design area does not exceed 10^{-7} per aircraft hour.

Further analysis of past accidents show that about 80% of the powerplant caused accidents can be attributable to the basic engine, the remainder being due to fuel supply failures, cowlings and other installational items. The aim for individual engine failures can then be assessed as follows.

Aircraft design limit for airworthiness catastrophic failures (all powerplant reasons)	10^{-7} (per aircraft hour)

Proportion for basic engine (four-fifths of the total powerplant risk)	8×10^{-8} (engine caused catastrophe per aircraft hour)
Proportion for each engine (assuming 4 engines per aircraft)	2×10^{-8} (engine caused catastrophe per engine hour)

At this point, we must pause and remember that in the early stages of engine design when the designer is attempting to aseess whether his design will meet the required limit, it is quite possible that the types of aircraft in which the engine may be installed will not be known. Certainly precise details of the installation will not be known. So whereas the designer can judge that certain events, eg serious non-containment, would be hazardous to the aircraft, he could not say with any certainty that it would be catastrophic. We are fortunately, for our peace of mind, dealing with small numbers when considering catastrophic accidents so that statistics are not much help, but we believe there is some justification for a broad assumption that only one hazardous failure in ten is likely to prove catastrophic to the aircraft. A further assumption is that there are about ten different failure modes which could produce hazardous effects. There are probably only three or four types of engine failure likely to hazard an aircraft, namely, significant non-containment of engine debris, fire, production of unacceptable toxic products and serious loss of engine control, eg inadvertent reversal of thrust, but there could be several modes of failure leading to any of these results. For example, failure of a disc could be initiated by overspeeding due to control system or shaft failure, or overheating due to cooling air failure.

So to continue with our arithmetic:-

Assuming that of every 10 hazardous failures, one will become catastrophic	2×10^{-7} (engine hazardous failures per engine hour)
Assuming 10 different failure modes which can produce a hazardous failure	2×10^{-8} (engine hazardous failures per engine hour).

Since there will almost certainly be failures not envisaged when carrying out a failure analysis, it seems reasonable that the aim should be that no hazardous failure of an engine from any individual failure mode should be predicted to occur more frequently than one in 10^8 engine hours and this is so stated in the British Civil Airworthiness Requirements. It should perhaps be added that the requirements also recognise that in dealing with probabilities of this lower order of magnitude, absolute proof is not possible and some reliance must be placed on engineering judgement and previous experience combined with a sound design and test philosophy.

LOSS OF THRUST IN FLIGHT

It is a basic requirement for any multi-engined civil aircraft that it must have acceptable performance with one engine inoperative. However, this cannot necessarily be extended to more than one engine for obvious reasons if the particular aircraft under consideration happens to be twin-engined. So when considering the required powerplant reliability target under this heading, it is the probability of having to shut down more than one engine on any one flight that must be explored and since the twin is typically the most vulnerable, it is the target for this type of aircraft that is considered in this paper.

The basic simplified formula for double engine failure is -

$$P_2 = \frac{n\,(n-1)}{2}\, p^2 \quad t^2$$

where P_2 = the probability of 2 engines having failed

n = the number of engines installed in the aircraft

p = the probability of engine failure/engine hour

t = intended flight duration

and this reduces to $P_2 = p^2\, t^2$ for a twin engine aircraft

or $P_2 = \frac{P^2 t^2}{2}$ if the aircraft returns to base following the first shutdown.

This now has to be modified further to take account of variations in flight procedure and variations of predicted probability over varying flight lengths. In addition there is perhaps a natural reluctance of a pilot to shut down a second engine and there is always the possibility that a shut-down engine can be relit. Consequently we have seen fit to introduce a 'failure/shut-down' ratio, r, to modify the actual in-flight shut-down rate.

The following assumptions based as far as possible on available evidence, have therefore been made:-

1 (a) If the flight duration is one hour or less the aircraft will carry on to its destination.

(b) For flights between 1 and 3 hours, the aircraft is assumed to turn back if engine shut-down occurs before the half-way point.

(c) Flights over 3 hours are required to have a diversionary airfield within 1½ hours following the first shut-down (ie have a mean time of flight after failure of 3/4 hour).

2 The failure/shut-down ratio, r = 0.5.

3 The in-flight shut-down probability, p, is the overall predicted rate applicable to a 2 hour flight. For a one-hour flight it is factored by 1.3, for a 3 hour flight it is factored by 0.9.

Using these assumptions -

For a 1 hr flight,	$P_2 = (1.3\ prt)^2$	$= 0.42\ p^2$
" " 2 " "	$P_2 = 0.5(prt)^2$	$= 0.5\ p^2$
" " 3 " "	$P_2 = 0.5(0.9\ prt)^2$	$= 0.91\ p^2$
" " 4 " "	$P_2 = 2(0.9\ pr)^2 \times 0.75 \times 4$	$= 1.2\ p^2$
" " 5 " "	$P_2 = 2(0.9\ pr)^2 \times 0.75 \times 5$	$= 1.5\ p^2$

and the results for various values of p are given in Fig 1.

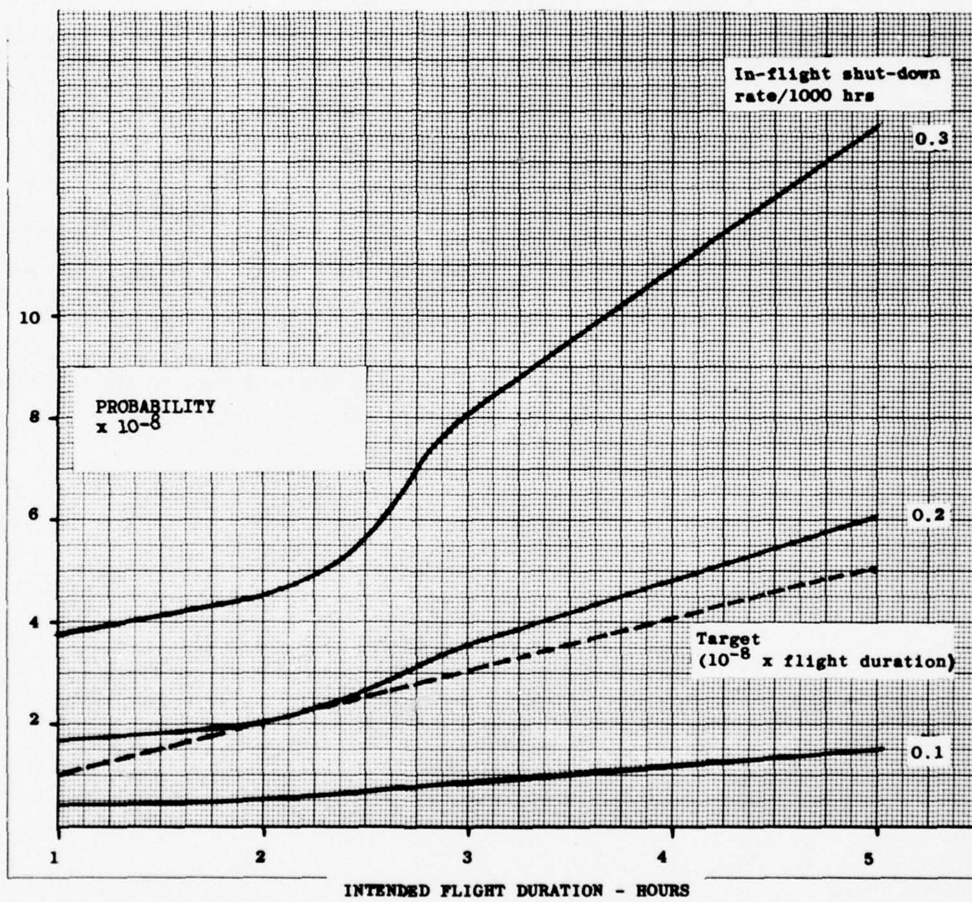


FIG 1

Probability of complete power loss in twin-engined aeroplane.

Bearing in mind our conclusions on overall airworthiness limits, it seems reasonable that the predicted probability of a fatal accident due to a power loss resulting from independent engine failures should not exceed 10^{-8} per hour of flight. Consequently it can be seen from Fig 1 that the probability of engine failure per 1000 hours should lie between 0.1 and 0.2, ie around 10^{-4} per engine hour, in order to achieve an acceptable level of airworthiness.

OVERALL RELIABILITY

The significance of overall powerplant reliability in terms of adequate engine performance and response at all times is more difficult to pinpoint than the two preceding objectives because there should be no hazard to the aircraft unless deterioration had been overlooked to such an extent that all engines were affected. Nevertheless, correct engine performance and response is important for the safety of the aircraft as even minor engine misbehaviour could increase the workload of the pilot, not to mention the effects on his nervous system, and this in itself could be contributory to an accident. However, in this case, the Airworthiness Authorities do not attempt to define a numerical limit but rely on design requirements applied at the time of engine certification followed by controlled maintenance once the engine goes into service. I am sure there are people here far more expert than I am in the field of maintenance so I do not intend to dwell very long on that particular subject. The old concept of a fixed overhaul period is rapidly giving way to maintenance as and when required, but it must be pointed out that the latter does require fairly sophisticated bookkeeping and may not be suitable for small operators. In any case critical parts, ie those parts the failure of which could hazard the aeroplane, will continue to require special attention and in most cases this will take the form of a fixed life.

The majority of engines in service have some form of thrust measurement indicated on the flight deck so monitoring engine performance is no problem. Various ways of engine health monitoring have been proposed and are used in service, but generally these have more impact on the economy of operation rather than airworthiness. All the Airworthiness Authorities can do is to try to keep abreast of the various ideas and ensure by monitoring service defects and failures that an acceptable level of airworthiness is being maintained.

CONCLUSION

Airworthiness is not a static art and must always take account of changes in the aviation world and consequently airworthiness targets are liable to change over the course of years although this is usually a very gradual process. I believe the sort of targets outlined in this paper for powerplant reliability are achievable and should provide the necessary yardstick against which reliability can be measured for many years to come.

I am grateful to the CAA for permission to present this paper and my colleagues for their help in preparing it. The views above are my own and should not be taken to represent those of the Authority.

DISCUSSION

J.C.Ripoll

Which are, in your opinion, the best reliability performance indications? I am worried by reading at the same time that

(1) the in-flight shut down rate per million flight hours is very low

(2) for a group of companies, in 1974 with 2000 jet engines in service, there were 2000 in-flight shut downs – with 200 turbo props in service there were 400.

The last figures are rather threatening in my sense. Moreover, shut down rate during take off is not the same as during cruise.

Author's Reply

Considering the curve of in-flight shut down rate with service experience (Figure attached) you can see in general that the turboprop and the large turbofans have started off worse than the straight jet engine. So it does appear from this somewhat limited information that turboprops and turbofans, either because of their sophistication or because of their size do start off at a rather high rate. On the other hand, except in one case, they have come down in a very satisfactory way and have ended up within the target. I think from the airworthiness point of view the in-flight shut down rate is the reliability indicator that really matters. There is certainly a variation in the shut down rate during take-off as opposed to cruise but at the levels we are talking about, the probability of having an engine failure at take-off is still extremely small.

A.Mihail

Should the 10^{-7} rate of dangerous anomalies be considered as applying to both a turbojet and a turbofan?

Author's Reply

As I tried to indicate in my paper, our target for catastrophic failure is largely based on past experience of what the public is prepared to accept. As a consequence, we do not distinguish between turbojets, turbofans or turboprops in that context.

A.Mihail

If the affirmative, is the turboprop propeller a part of the powerplant unit (or should it be regarded as such)?

Author's Reply

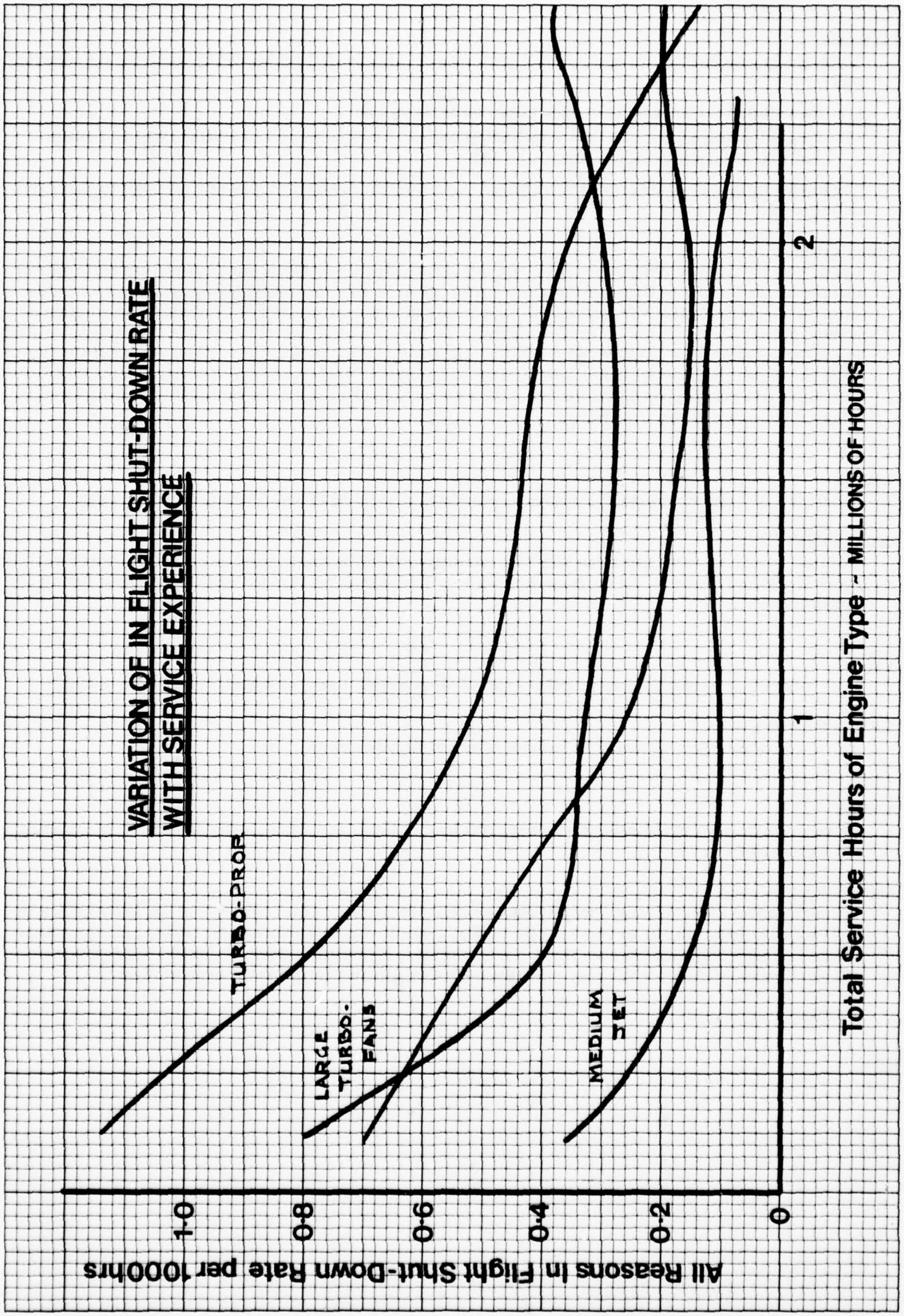
Yes, the propeller should obviously be included as part of the powerplant.

A.Mihail

In the paper, the rates of anomalies are expressed per hour of flight; in some cases, should they not be expressed per flight cycles?

Author's Reply

I admit that I have tried to dodge that particular problem. In my paper I have talked about one in ten million aircraft hours or aircraft flights, so I have left the options open. If you are considering the landing gear, for example, something that is only used twice per flight, it is reasonable to consider it in terms of aircraft flights. If you are talking of engine operation which goes on continuously then I think it is right to use hours of flight.



VARIATION OF IN FLIGHT SHUT-DOWN RATE
WITH SERVICE EXPERIENCE
TURBO-PROP
LARGE TURBO-FANS
MEDIUM JET
All Reasons In Flight Shut-Down Rate per 1000hrs
1·0
0·8
0·6
0·4
0·2
0
1
2
Total Service Hours of Engine Type - MILLIONS OF HOURS

RELIABILITY VERSUS COST IN OPERATING WIDE BODY JET ENGINES

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SUMMARY

Scheduled international airline operational and related maintenance characteristics require a high degree of reliability from aircraft and engines to be able to offer a safe but also commercially and economically justified product. This required high degree of reliability, is ultimately gained and maintained at high cost to the airline. Although for a specific engine type and a specific airline, theoretically, an optimal cost/reliability level to be reached should exist, practically this cannot be found due to unquantifiable variables. The only way to reach a justified cost/reliability level, apart from initial engine design, is to look for cost effectiveness of offered product improvements, changes in maintenance organization/philosophy/practices and improvement of monitoring and control techniques. Each of these points is discussed in more detail and based on experience gained by operating the new generation of high performance high by-pass jet engines.

PREFACE

With the appearance of the wide body aircraft a new class of jet engines entered service namely the high power high bypass ratio jet engines. These engines are characterized by an increase of scale, incorporation of advanced technology and design features based on new ideas with respect to maintenance.

The wide body aircraft and their engines faced the airlines with financial, organizational and technical problems to solve. In Europe this resulted in the establishment of two consortia namely ATLAS and KSSU, each consisting of a number of European airlines, with the objective to most economically perform the engineering and maintenance work on the wide body aircraft and their engines. Within the KSSU consortium KLM is responsible for engineering and maintenance of the General Electric CF6 engines. Therefore this presentation on reliability versus cost will deal with the experience gained in operating this type of engine.

One cannot expect anybody to give a complete picture of the relationship between reliability and cost for a certain engine type. Reliability a.o. is influenced by inherent design deficiencies, operating environment and maintenance policy while cost consequences a.o. are depending on the kind of airline operation, the way an airline is organized, the scale of operation and the airline standards. While these factors differ for each airline the only thing which can be done is to point out important items which show to have an impact on cost and also show to have potential for cost effective improvement. This is what this paper will deal with, without having the objective to be exhaustive.

RELIABILITY. WHAT DOES IT MEAN

Commonly reliability is defined as the probability that an item will perform a required function under specific conditions without failure for a specific period of time. This is a general definition which forms a guideline to establish parameters by which one can measure reliability of a certain product or individual components. Speaking about aircraft engines failing to meet the propulsive function results a.o. in pilot complaints, aircraft delays, inflight shut downs and engine removals. Therefore measuring reliability can be done by defining parameters like the number of these kind of occurrences within a suitable defined time related interval. One then gets generally used reliability parameters like the engine removal rate being the number of engine removals per thousand (1000) engine running hours and the engine inflight shut down rate being the number of inflight shut downs per thousand (1000) engine running hours.

Although these parameters mentioned indicate the reliability of the engine as a whole due to their operational character they only cover part of the total picture. This total picture shall include the more detailed information about units or parts of the engine responsible for the removal or inflight shut down expressed in terms of the number of occurrences per time period. Moreover it has to be mentioned that the various parameters are not covering independent occurrences. For instance an inflight shut down comes to be noted as a pilot complaint and can result in an engine removal which in turn can lead to an aircraft delay. It will be clear that reliability when used in connection with so complex an item like an aircraft engine has a wide scope and that an overall reliability figure does not exist.

WHAT DOES AN AIRLINE REQUIRE

Airlines by their very nature have to fly and make a profit by it. Waisting ground time means ineffective use of a rather expensive means of production which is especially true for the present generation of wide body aircraft. International scheduled airlines have to fly a published schedule on which their customers should be able to rely. Characteristics of scheduled international airline operation therefore are high aircraft utilization, short turnaround times in between flights, punctual operation and most of the aircraft movements taking place away from home base. These characteristics apart from the safety aspect require the highest possible degree of reliability from aircraft and systems from which the propulsion system is a very important one. Important reliability parameters are delay rate, inflight shut down rate and removal rate. Moreover this

high degree of reliability should be reached at minimal cost, whereby things like engine failure modes (costly secondary damage) and rate of unscheduled removals (costly outer station removals, inconvenient removal times) play an important role. For instance an engine failure at an outer station means a.o. having to transfer passengers to other airlines or sending a relief aircraft, bringing in from the main base personnel and equipment together with a serviceable engine to return the aircraft into service again. Another example is an engine failure early in climb, this normally means having to dump a considerable amount of fuel which is rather expensive nowadays, to get from take off weight to allowed landing weight. Typical cost figures for a DC-10 operation are displayed in tabel shown in figure 1.

TYPICAL DC-10 COST DATA

Situation	Cost	Remarks
Outer station removal	Minimum $ 100,000.-	Loss of 3 - 4 airplane days. Airplane value $ 26 million.
Engine shut down in climb resulting in fuel dump.	Minimum $ 10,000.-	Only accounts for fuel dumped. Cost of aircraft delay and resultant cost excluded.

Figure 1

Apart from this there is the passengers discomfort which works negatively on the company's image if things like this occurs frequently. If reliability would be low and therefore the chance of both examples to occur regularly would be high, this to an airline would mean loss of income not only on short term but also on the long term due to degrading product quality and this together with high expenses works the wrong way for making a profit.

HOW DOES RELIABILITY RELATE TO COST

As has been said before an airline requires a high degree of reliability from its engines but of course not to be gained at any cost. One has to minimize total cost composed of the cost of achieving and maintaining a certain reliability level and the resultant cost of having such a level. Figure 2 displays total cost as resulting from these two cost factors.

COST/RELIABILITY RELATIONSHIP

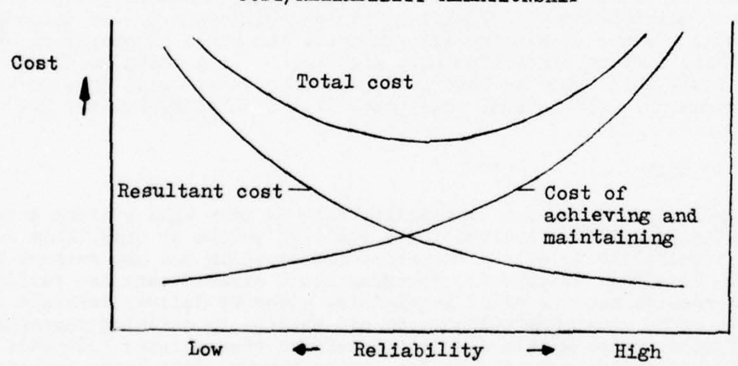

Figure 2

The resultant cost a.o. is composed of aircraft delay cost, inflight shut down related cost (fuel dump), cost of relieving stranded aircraft and passengers and maintenance related cost (material consumption, repair related labour cost, inventory cost). This resultant cost most probably will decrease with increased reliability. The cost to achieve and maintain a certain reliability level a.o. is composed of monitoring cost, product improvement introduction cost (hardware, labour, campaign cost) and increased workshop inspection and related repair activity. This cost in general will increase with a higher reliability level to be reached and maintained. There are a number of things to be said about these two cost factors. First the relationship of both factors with reliability differs from airline to airline due to different operational conditions, company organization, standards and policies and so on. Second the level of the achieving and maintaining cost curve greatly depends on maintenance organization/concept and practices. This for instance means that a lower cost level can be reached at the same reliability level by changing towards a more efficient set up. Third a great number of the elements from which these cost are composed are nearly if not at all to quantify. Forth, one can not establish the total "achieving and maintaining" cost curve at a certain moment in time while future improvement possibilities and their reliability upgrade potential are unknown and therefore no cost can be assigned to them.
This means that a generally valid optimal reliability level for all airlines does not exist and secondly that due to the many unquantifiable and unpredictable cost elements the curves can not be established. Practically speaking finding the way to the optimum reliability/cost level is a

step by step approach, at each step weighing the cost of an available improvement against its benefits maintenance and operational wise and moreover always being alert for ways to change the overall cost level without loss of reliability.

WHAT ARE THE IMPORTANT ELEMENTS IN ACHIEVING AN OPTIMAL COST/RELIABILITY LEVEL

In the process of achieving an optimal cost/reliability level in a step by step way one has to focus attention on a few areas of importance which are:

- Introduction of product improvements.
- Monitoring and control techniques.
- Maintenance organization/concept/practices.

Each of these areas will be discussed in more detail in the next paragraphs.

INTRODUCTION OF PRODUCT IMPROVEMENTS

We have to accept the fact that a perfect product can not be designed initially. This a.o. is due to the designer not being able to fully envisage the real environment in which and moreover the way by which his product will be operated. Also elaborate testing of the product before it enters service is not revealing all it should be. As a result of this airlines have to accept deficiencies and weak spots to show up during operation ultimately resulting in product improvements offered by the manufacturers. It will be clear that the fewer the necessary product improvements to be introduced, and if they have to the soonest they are offered by the manufacturer, the better off the airlines will be. Figure 3 shows the yearly introduced service bulletins (product improvements) and the number accepted by the airline from the introduction of the CF6-50 engine on for the KSSU consortium.

OFFERED/ACCEPTED PRODUCT IMPROVEMENTS

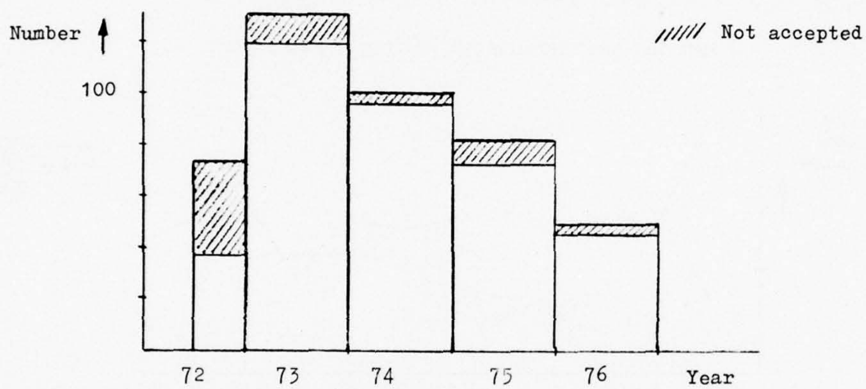

Figure 3

So large a number of modifications to be performed requires a rigid priority system to be effective. To decide upon incorporation of a certain modification but introducing it slow means waisting money. One never can introduce all modifications during a certain shop visit while this normally means opening up the complete engine. Tabel 4 shows the break down in applied priorities and its average percentage of all available modifications for each priority.

MODIFICATION PRIORITIES AND RELATIVE DISTRIBUTION

Priority	Percentage	Remarks
1	1	Mandatory campaign.
2A	60	Mandatory during shop visit on affected/ non affected modules.
2B	15	Mandatory at shop visit on affected modules. only
3	25	At exposure during shop visit.
4	0	Based upon parts condition.

Figure 4

Product improvements are directed towards solving design deficiencies, improve deterioration characteristics and extend part life capability. In making the trade off calculation needed to decide whether to accept a product improvement yes or no one has to take these considerations into account to recognize the relevant cost elements. Also the priority to be given to a certain modification will depend on this.

The effect of such an approach will be that the fleet which normally is a mixture of old and new engines gradually will converge to the cost/hour related to the "maturity" level of reliability as is depicted in figure 5.

PRODUCT IMPROVEMENT INTRODUCTION EFFECT

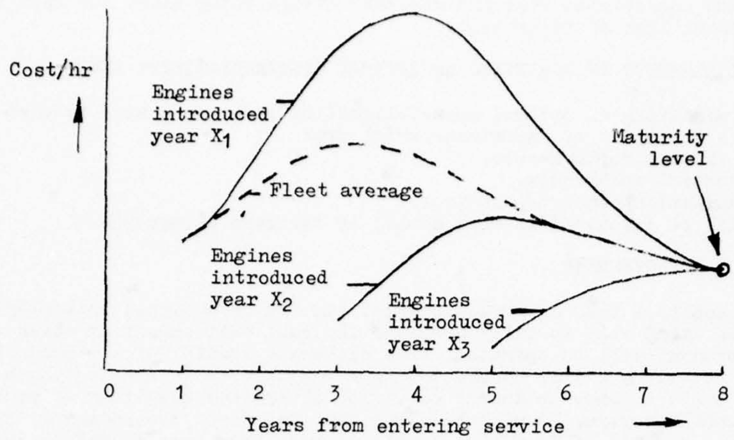

Figure 5

Figure 6 shows the actual "manhour consumption per engine running hour" curves for the CF6-50 engine up till now as compared to the JT3 engine.

MANHOUR CONSUMPTION PER ENGINE RUNNING HOUR

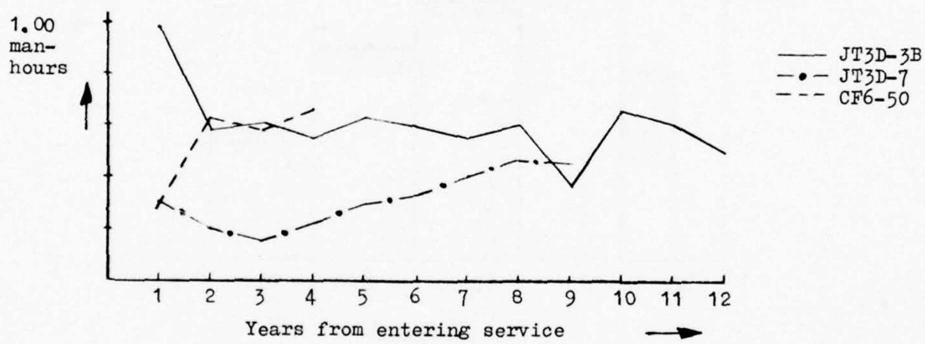

Figure 6

MONITORING AND CONTROL TECHNIQUES

The wide body jet engines were introduced to start service on the basis of the "on condition" concept. This concept requires having an "on wing" condition monitoring system together with a maintenance inspection program. Also when an engine is in the shop forced and opportunity inspections are to be performed on individual parts to access condition and continued life potential. The appropriate monitoring and control techniques cover a variety of things like visual checks, metal checks, X-ray, boroscope inspections, spectographic oil analysis, trend charts, statistical surveys and use of AIDS. All these things are essential to control reliability but thereby also cost. Especially with the present generation of high power jets operating at high pressure ratios and temperatures, involving latest but by that costly technology, early detection of potential failure modes is necessary to avoid expensive secondary damages. Tabel shown in figure 7 gives an impression of dollar value involved in secondary damages as compared to correcting the primary failure mode alone.

REPAIR COST COMPARISON FOR PRIMARY AND SECONDARY DAMAGE

Failure mode	Primary repair cost	Primary + secondary repair cost
Mini nozzle bolt	$ 75,000.-	$ 430,000.-
Turbine blade HPT	$ 110,000.-	$ 360,000.-
St. 2 blade retainer HPC	$ 80,000.-	$ 150,000.-
St. 7 HPC blade	$ 24,000.-	$ 120,000.-

Figure 7

Apart from preventing secondary damage also trying to avoid outer station removals is an objective. An outer station removal at least costs about $ 100,000.- if this occurs.
Figure 8 shows the effect of the at present most important monitoring and control techniques which indicates boroscoping to be the most effective technique. Although one has to keep in mind that background knowledge of how far to go if a defect is detected (for instance a crack) is a very important thing but difficult to acquire.

EFFECT OF MONITORING & CONTROL TECHNIQUES ON UNSCHEDULED REMOVALS

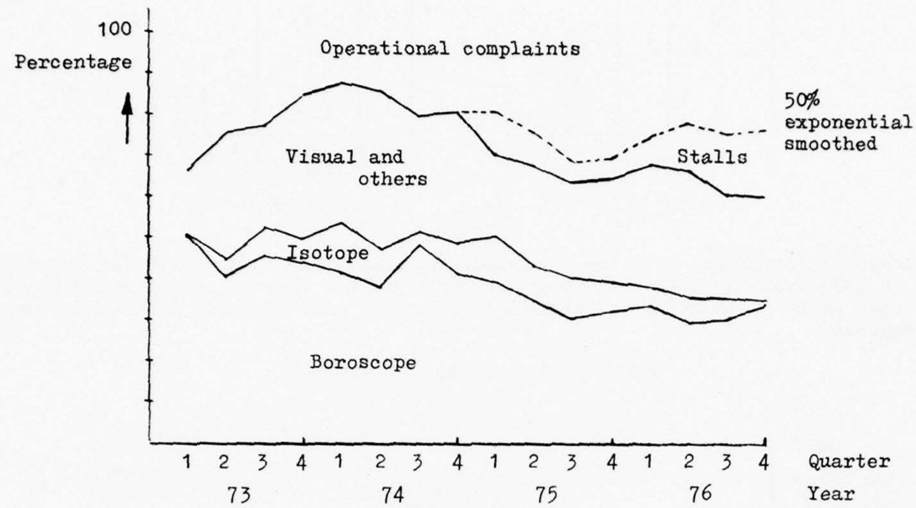

Figure 8

For illustration purposes a relatively new problem area to the CF6-50 engine is depicted, namely stall which tends to decrease the effectivity of the monitoring and control techniques. A relatively new technique offered with the wide body aircraft is AIDS meaning Airborne Integrated Data System which in essence shows to have a great potential in early failure detection. Unfortunately this technique due to its present state of the art only serves as a backup information source while it can be a powerful monitoring instrument to indicate engine condition. This requires heavier engine instrumentation and memory capacity but could be worthwhile in trying to bring fault isolation "in flight" instead of in the limited ground time and moreover improve the quality of the condition information.

MAINTENANCE ORGANIZATION/CONCEPT/PRACTICES

It is obvious that even the best product if not taken care of properly will degrade and thereby starting to be more expensive than necessary. It is the way one organizes maintenance the framework in which one performs it and how one actually does it which to a great extend determines the cost level and the reliability of the product maintained. Especially the maintenance concept turns out to be of prime importance. The wide body jet engines as has been said before were introduced with an "on condition" concept but moreover modularity as a concept was introduced. This modularity when used fully enables an airline to minimize shop visit time, thereby greatly influencing investments in spare engines but also repair and storage facilities. There still is another important factor involved which is vital to the "on condition" concept namely workscope planning. To be cost effective and to improve or at least prevent reduction of reliability one has to accomplish a workscope at each shop visit which optimizes cost versus "on wing" time and not only limit oneself to just fixing the failure or problem. To establish this workscope one has to take into account the previously discussed product improvements available, the data generated by monitoring the fleet, shop findings, parts sampling data, engine deterioration data and analyse all this to find out which activities are necessary to achieve a preset "time on wing" objective. From this a "minimum" work standard for each shop visit can be defined which is enlarged for the individual engine based on this specific engine's health determined by a variety of data like trend data, vibration trends, spectografic oil analysis, boroscope findings and so on and the history of modules the engine is composed of. The decision as to the final workscope for a specific engine within the KLM organization is a team effort within a committee formed by representatives of engineering, quality control, production and production planning. The effectivity of this kind of workscope planning can be measured by a "time since last shop visit" distribution, together with an analysis of the reasons for low time removals. Figure 9 shows two successive distributions covering first and second half of the year 1976. From this it can be seen that a shift towards the lower end has taken place indicating new problem areas resulting in low time removals. Analysis going on has to show where the workscope has to be adapted.

TIME SINCE LAST SHOP VISIT DISTRIBUTION

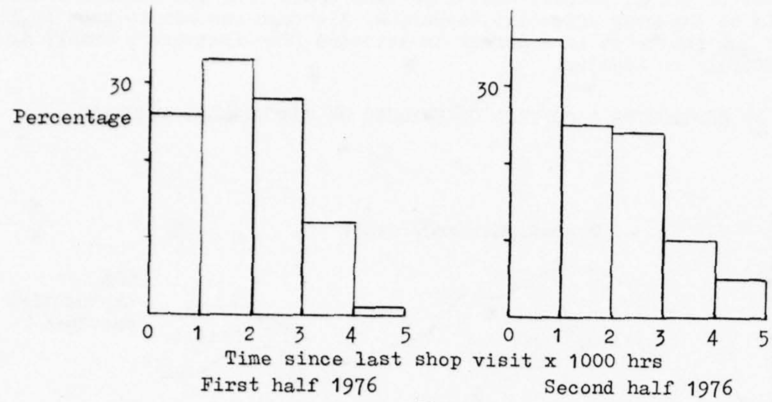

Figure 9

Figure 10 shows the development of the "average time since last shop visit" for the CF6-50 engines maintained at KSSU from the start of operation. This picture reveals an improving trend although quite some more improvement still is possible if only one could get rid of the high percentage of low time removals.

TIME SINCE LAST SHOP VISIT TREND

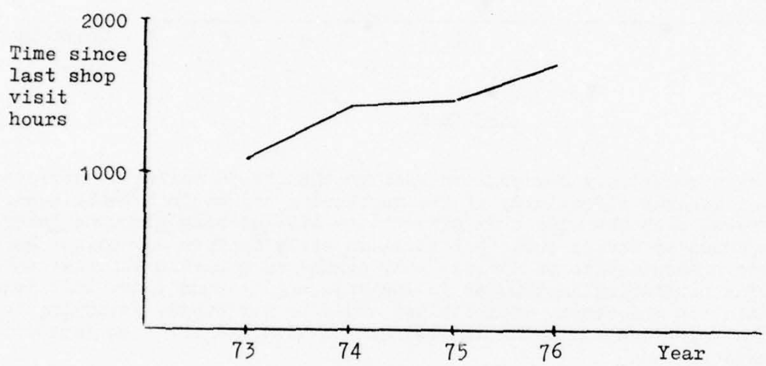

Figure 10

CONCLUDING REMARKS

Concluding this presentation it can be stated that a fixed relationship between reliability and cost does not exist, while this depends on many variables having a different impact with different airlines.
What is clear is that there is an optimal cost/reliability level which ultimately will be achieved if one keeps being aware of the potential within the mentioned elements: product improvement introduction, monitoring and control techniques and maintenance organization/concept/practices and more in particular cost effective workscoping.

DISCUSSION

A.Mihail

La figure 6 montre que le cout main d'oeuvre du CF 6 (réacteur de type modulaire de la dernière génération) est du même ordre que pour les moteurs première génération (JT4, JT3D). Cela est surprenant après tout ce qui a été dit, avant la mise en oeuvre, sur le prix d'exploitation de ce type de moteur. Qu'en pensez-vous?

Author's Reply

It is of course very difficult to say whether your initial statement is correct. What you imply is that the high generation, that is the third generation of jet engines should be lower in manhours consumption with respect to running hours than the JT3 engine. It is a more complex engine, it is a more powerful engine and it applies higher technology, it applies new techniques, which could explain that it needs more manhours. But it shows actually to be compatible with the JT3.

A.Mihail

Votre presentation parle peu des AIDS. Or KSSU est le groupe qui a admis des dépenses importantes sur ce système. Estimez-vous que les faibles résultats obtenus, d'après votre exposé, sont dus en fait à la faible fiabilité du système ou à sa conception?

Author's Reply

You are right that we did spend quite a lot of money in introducing the AIDS system into our fleet. And if I had to address to why we are not fully content with this system, I can say it has indeed two elements which you have already mentioned. One was the very poor reliability it had in the beginning. It is improving now.

But the second one is the concept that is of course something we have in hand. In our normal operation it is only used to find out after a failure has taken place, how this actually developed. It can give some indication where to look at for the start of the failure. Therefore, I said it is used as back up information. It could be a valuable instrument if you could monitor the engine and at least could keep a record of many flights in its memory. There you could really do some trend analysis, but that is far future. I guess if you were able to achieve gas path analysis, you could already during operation indicate where something is degrading within the engine, that is what I mean when saying it could be more effectively used. Of course that would require additional engine instrumentation as well as enlarged memory capacity of the computer.

LES RISQUES AFFECTANT LA RESISTANCE STRUCTURALE

ET LA SECURITE DES PROPULSEURS MODERNES

par
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RESUME

L'évolution des performances des propulseurs modernes déjà en service ou qui le seront dans les années 1980 et au-delà nécessite la mise en oeuvre de caractéristiques de plus en plus avancées en matière de niveaux de température, d'efforts et de contraintes mécaniques et thermiques et de nouvelles technologies de matériaux afin d'en améliorer les rendements. Une attention accrue est par suite fondamentalement nécessaire au stade de la conception et du développement pour préserver l'intégrité structurale de ces propulseurs. La présentation s'attache à l'analyse des risques auxquels l'intégrité structurale de tels propulseurs est soumise, compte tenu de l'expérience importante déjà accumulée et dans la perspective des niveaux d'énergie thermique ou cinétique mis en oeuvre à l'avenir. Des exemples précis sont discutés à cet égard pour justifier la nécessité de telles précautions dans les domaines cinétique (containement) et thermique (emploi du titane, etc...)

Plusieurs années d'expérience dans le transport aérien international avec les propulseurs à taux de dilution élevé mis en service successivement depuis le début des années 1970 (Pratt & Withney JT9D et General Electric CF6) sur les avions à grande capacité (B-747, DC10-30, A300B) a permis progressivement de mesurer et d'apprécier la nature l'importance et les conséquences techniques ou économiques de l'ensemble des problèmes rencontrés dans l'utilisation de ces moteurs de conception avancée.

Certains de ces problèmes ont présenté néanmoins une dimension toute particulière par les risques potentiels sérieux directs ou indirects qu'ils comportent sur l'intégrité structurale et sur la sécurité de fonctionnement de ces réacteurs et dont on peut estimer qu'il est par suite essentiel du point de vue d'un utilisateur d'en effectuer une analyse approfondie pour définir les domaines où des programmes de développement paraissent nécessaires pour leur apporter des solutions efficaces.

Un tel examen revêt au surplus, un intérêt encore plus grand dans la perspective des programmes actuellement engagés pour le développement de la seconde génération de réacteurs à taux de dilution élevé (CFM-56 et JT10D) et sur lesquels des problèmes analogues seraient tout à fait susceptibles de se produire si l'expérience déjà obtenue n'était pas utilisée d'une manière suffisante en temps voulu.

Bien que l'on puisse reconnaître volontiers que l'analyse qui suit soit plus aisée après une expérience étendue plutôt qu'au stade initial du développement, on peut estimer néanmoins qu'elle reste essentielle pour aider à améliorer la conception des propulseurs modernes existants et à venir.

Le tableau 1 résume la nature des risques majeurs qui ont affecté à des degrés divers l'intégrité structurale des réacteurs à taux de dilution élevé. Ces risques en dernière analyse résultent essentiellement des niveaux d'énergie thermique et cinétique particulièrement élevés mis en oeuvre dans le fonctionnement thermodynamique ou mécanique de ces réacteurs.

Ces risques en soi ne sont pas nouveaux et ont figuré au premier plan des préoccupations des constructeurs, des compagnies utilisatrices et des autorités de certification dès la mise en service des propulseurs à réaction des générations antérieures depuis le début des années 1960. Néanmoins, les capacités présentes d'avions tels que les Boeing 747, Douglas DC 10-30 et Airbus A300B ou celles des avions futurs, leur donnent et continueront à leur donner une bien plus grande dimension que les avions précédents.

L'importance des niveaux plus élevés d'énergie thermique ou cinétique dans certains modes d'incidents, découlent en effet, de l'évolution de certaines caractéristiques particulières à la conception et à la mise en oeuvre des réacteurs à taux de dilution élevé dans plusieurs domaines fondamentaux :

- Performances
- Cycle thermodynamique
- Conception basique
- Technologie
- Politique d'entretien en service.

Pour analyser en détail les processus pouvant affecter l'intégrité structurale de ces réacteurs, il a paru commode de rassembler et d'utiliser les données générales figurant dans le tableau 2.

On a recherché dans ce tableau à classer les risques structuraux en fonction de leur origine, thermique ou cinétique, et à évaluer pour chaque origine dans quelle mesure la sévérité de ces risques particuliers varient ou pourraient varier en fonction d'une variation indépendante de caractéristiques fondamentales de conception ou d'utilisation de ces réacteurs.

Ce tableau comporte nécessairement dans certains domaines marginaux une part d'appréciation inhérente à son caractère de généralisation et en dépit du fait qu'il soit basé uniquement sur des considérations d'expérience.

Néanmoins on peut considérer que ces données permettent de schématiser les domaines les plus critiques, et de souligner plusieurs points particulièrement importants au stade de la conception qui requièrent une attention toute spéciale.

(1) RISQUES STRUCTURAUX D'ORIGINE THERMIQUE

(1.1) Carburant

Les fuites de carburant en général, internes ou externes, comportent potentiellement une augmentation importante de niveaux de risques sous l'effet séparé ou combiné des facteurs suivants, qu'il est facile de souligner par quelques exemples :

(a) Performances
. Augmentation de la poussée de décollage
. Augmentation du rapport poussée/poids
. Rendement du cycle thermodynamique.

(b) Conception basique
. Gradients de température.

l'augmentation de poussée entraîne celle du débit de carburant et donc, a priori, de l'importance des fuites susceptibles de se produire dont les conséquences se trouvent d'ailleurs aggravées en cas de système de draînage insuffisant comme l'expérience l'a montré.

L'augmentation du rapport poussée/poids a coïncidé dans un cas typique avec un allègement des injecteurs de carburant dont l'expérience a révélé les conséquences fâcheuses sur leur résistance à la cokéfaction et leurs limites de fatigue haute fréquence puisque des criques ont pu se développer jusqu'à une fracture locale avec fuite de carburant interne, combustion incontrôlée, finalement perforation locale du carter diffuseur des réacteurs intéressés et incendie localisé du réacteur et des équipements adjacents (figure 1,2 et 3).

Les gradients de température se produisant dans ces injecteurs ont été trouvés ultérieurement anormalement élevés au cours d'essais sur plates-formes ou sur réacteurs complets. Cette catégorie d'incidents à nécessité le développement d'un nouvel injecteur renforcé et amélioré dans lequel les gradients externes de température ont été réduits de 297°F dans l'injecteur initial à 46°F dans l'injecteur amélioré grâce à une étude plus soignée (figure 4).

Quant au rendement du cycle thermodynamique, son augmentation entraînant celles des températures de combustion, on conçoit, et l'expérience a confirmé, qu'il en résultait bien une élévation de l'environnement thermique du réacteur susceptible d'aggraver à son tour l'effet individuel ou combiné des facteurs précédents, notamment sur l'ensemble du circuit de carburant et de ses connexions de raccordement multiples pour des facilités d'entretien modulaire.

(1.2) Lubrifiant

L'expérience a montré que les risques de feu interne d'huile peuvent être sensiblement augmentés sous l'effet de plusieurs facteurs.

(a) Cycle thermodynamique
. Augmentation du taux de compression
. Augmentation de débit d'air.

(b) Technologie
. Utilisation de joints d'étanchéité de compresseur à labyrinthes.

Un exemple typique illustrant ce genre de problème et le rôle des facteurs précédents est celui du circuit d'air de ventilation du compartiment de roulement principal n° 3 d'un réacteur à taux de dilution élevé. Ce roulement à rouleaux est situé en position arrière du compresseur haute pression et son compartiment est isolé par un système complexe de joints d'étanchéité à labyrinthes destiné à empêcher l'air sortant du compresseur haute pression à température élevée de pénétrer dans ce compartiment, c'est-à-dire d'y provoquer la cokéfaction de l'huile.

Un équilibre de pressions d'isolement est ainsi réalisé, mais qui, en particulier avec l'usure mécanique des labyrinthes et l'accroissement de leurs jeux permet effectivement l'admission d'air haute pression dans ce compartiment du fait du taux de compression lui-même très élevé du compresseur. La température de cet air (530°C) étant très supérieure à celle du point d'auto-inflammation de l'huile synthétique employée (260°C), il en résulte la combustion locale de l'huile entre le compartiment du roulement n° 3 et la boîte d'entraînement d'accessoires à laquelle le compartiment est ventilé et le déclenchement d'un incident d'huile violent dans cette boîte entraînant sa destruction partielle et l'établissement d'un feu d'huile plus ou moins impossible à arrêter entre temps, tant que le réacteur tourne même en moulinet, c'est-à-dire tant qu'un débit d'air et d'huile suffisant continuent d'arriver dans le compartiment du roulement (figure 5). Ce processus s'est effectivement produit dans un certain nombre de cas avec des dégâts secondaires très importants.

Cette catégorie d'incidents à justifié le déclenchement de mesures contraignantes en entretien pour surveiller l'évolution de la marge de température de l'air de ventilation du roulement 3 par rapport à certaines limites fixes afin de déceler à l'avance ce genre d'anomalies.

Néanmoins, on peut retenir comme conclusion de cette expérience que le principe d'utiliser pour l'étanchéité d'un compartiment de roulement un débit d'air à une température franchement plus élevée que le point d'auto-allumage de l'huile du compartiment à isoler est en soi une disposition qu'il est assurement préférable d'éviter.

(1.3) Titane

Le tableau 2 montre assez clairement que le risque constitué par les feux de titane devient particulièrement grave avec l'augmentation des valeurs d'un assez grand nombre de paramètres intéressant les performances, le cycle thermodynamique, etc... c'est-à-dire avec le développement de réacteurs de performances de plus en plus avancées et lorsque le titane continue d'être employé parallèlement de manière particulièrement intensive dans les ailettes, les aubes, les disques et les carters de compresseur en raison de ses caractéristiques mécaniques remarquablement intéressantes.

Au terme d'années d'utilisation de ce métal dans des réacteurs civils à taux de dilution élevé et même dans certains réacteurs des générations antérieures, l'expérience a montré que le titane comportait trois inconvénients majeurs :

(a) Ses limites de fatigue cyclique , par conséquent les risques de fractures en fatigue haute ou basse fréquence sont considérablement affectées par les effets d'entailles mécaniques et les discontinuités métallurgiques structurales internes ou superficielles.

(b) Le titane donne lieu à une combustion exothermique à tendance explosive à température très élevée (2000°C) lorsque les conditions appropriées de débit, de pression et de température d'air élevées sont réunies.

(c) Le titane possède un coefficient de conductibilité thermique qui n'est que 25 % de celui de l'acier environ. Il en résulte que le frottement mécanique titane/titane par suite de son faible transfert de chaleur se traduit par des températures superficielles extrêmement élevées susceptibles, dans les conditions précédentes et avec la génération appropriée de débris d'amorcer la réaction explosive de combustion.

Cette réaction est d'ailleurs extrêmement courte et est exprimée le plus souvent en milli-secondes.

Les premiers feux de titane sur réacteurs civils remontent à 1962 et l'importance exceptionnelle des dégâts secondaires souvent non contenus qu'ils entraînent et de leurs risques potentiels pour la sécurité de vol a été rapidement reconnue. La figure 6, montre un exemple récent assez caractéristique des dégâts internes résultant d'un feu de titane.

Ces incidents sont le plus généralement consécutifs à une rupture initiale d'ailette de rotor ou de stator compresseur mais leurs conséquences secondaires peuvent comporter la fusion de certaines tuyauteries d'huile ou de carburant avec les feux correspondants.

Quoiqu'il en soit, l'ensemble de l'expérience précédente permet d'estimer que l'utilisation du titane sans limitations ou restrictions telle qu'elle a été jusqu'à présent pratiquée, et en l'absence d'une technologie susceptible de réduire sérieusement les inconvénients majeurs précédents, malgré simultanément une évolution prévisible et importante de nombreuses caractéristiques de conception contribuant à les aggraver, ne saurait se prolonger sans qu'il en résulte une escalade redoutable dans les risques affectant l'intégrité structurale de certains réacteurs présents ou futurs, c'est-à-dire en définitive la sécurité du vol elle-même.

Pour ces raisons on peut également fermement souhaiter que la mise en oeuvre du titane devienne à l'avenir règlementairement contrôlée, de telle sorte que la résistance structurale au déclenchement, à la propagation et au minimum au containement des feux de titane devienne une partie intégrante des conditions de la certification des réacteurs futurs et à ce titre, fasse l'objet d'essais complets de substantation appropriée.

(2) RISQUES STRUCTURAUX D'ORIGINE CINETIQUE

(2.1) Disques

Les risques structuraux résultant des niveaux d'énergie cinétique considérables mis en oeuvre dans les disques sont nécessairement susceptibles d'augmenter potentiellement avec les facteurs suivants :

Performances : - Augmentation du rapport poussée/poids
- Poussée au décollage

Conception basique : Elévation des niveaux de contraintes

Technologie : Nouveaux matériaux

Il est juste de reconnaître que les progrès réalisés dans la conception, le développement et les essais en matière de disques de turbines ou de compresseurs ont été orientés jusqu'à présent avec l'objectif de leur assurer une intégrité structurale virtuellement totale. Néanmoins, l'expérience a montré que cet objectif n'a pas été entièrement atteint. En dépit du petit nombre de ruptures de disques, on peut estimer que les implications structurales pour le réacteur et l'avion d'une rupture non contenue de cette nature sont telles que des efforts supplémentaires soient essentiels pour assurer à ces disques une fiabilité encore meilleure.

Dans cette perspective, il est bon de rappeler que le rapport poussée/poids a pratiquement doublé en 15 ans sur les réacteurs militaires et augmenté de plus de 70 % sur les réacteurs civils. Ces progrès ont été possibles grâce à une augmentation des niveaux de contraintes tangentielles par l'introduc-

tion de matériaux nouveaux, de marges de contraintes plus réduites de conception et d'amélioration dans les techniques de refroidissement.

Néanmoins, malgré l'introduction de nouveaux matériaux ayant de meilleures propriétés de traction et de fluage, les caractéristiques liées à l'initiation et à la propagation des criques ne paraissent pas avoir suffisamment progressé. Il en résulte que des programmes de recherches importants restent encore à poursuivre tant par une meilleure analyse et une meilleure simulation des contraintes et des températures de disques résultant des profits de vols telles qu'elles sont mises en oeuvre dans des essais cycliques accélérés que par l'étude plus complète des caractéristiques métallurgiques et des limitations pratiques contrôlant les phénomènes de fatigue basse fréquence.

Dans ce dernier domaine certaines anomalies métallurgiques importantes ont pu être constatées récemment par exemple, telles que la présence de criques de fatigue dans un rotor de compresseur en titane Ti 6-2-4-2 à partir d'une zone contenant des décohésions et une forte ségrégation locale en phase alpha dure dans une matrice de phase beta ductile. En dépit des caractéristiques connues de cet alliage, l'anomalie en question a entraîné la rupture de ce rotor en cours de décollage.

La mise en oeuvre par ailleurs, de nouvelles techniques d'élaboration telles que celle du forgeage isostatique de disques à partir de poudres nécessitera des recherches approfondies pour contrôler les facteurs susceptibles d'intervenir dans les limites de fatigue basse fréquence de tels alliages tout en maintenant des limites de vie cyclique économiquement acceptables.

(2.2) Ailettes

Le tableau 2 montre que le nombre de paramètres contrôlant directement l'intégrité struturale des ailettes de compresseur en particulier, est très élevé. Au surplus, l'évolution prévisible de ces paramètres, dans le sens des caractéristiques des réacteurs à taux de dilution élevé déjà en service ou des réacteurs de la génération suivante, montre que les ailettes de compresseur continueront dans une large mesure à constituer une source interne de risques structuraux très importants dont il importe d'évaluer et de réduire les niveaux.

En se limitant seulement aux grandes lignes de l'expérience obtenue, on peut estimer essentiel que les ailettes de compresseur soient rendues d'une manière générale beaucoup moins sensibles à certaines conditions résultant de leur environnement sur des étages particuliers telles que l'importance et la nature du frottement à leur extrémité en raison de jeux initiaux particulièrement réduits pour des raisons de taux de compression élevé par étage, le pompage occasionnel pour des raisons d'altération de profils aérodynamiques ou d'anomalies occasionnelles de calages d'aubes de stator à incidence variable, les effets d'impacts divers sur les limites de fatigue etc...

L'expérience a dernièrement montré en particulier, combien la nature du frottement en bouts d'ailettes jouait par exemple un rôle important sur la structure du réacteur. Dans ce cas particulier, et à l'issue d'un programme d'essais ayant mis en oeuvre des moyens d'analyse et d'études sans précédents, il a été possible de vérifier et de reproduire au banc l'explosion du carter haute pression sur plusieurs réacteurs à la suite de la génération par usure consécutive au frottement et au deséquilibrage du rotor, du revêtement epoxy du carter compresseur à certains étages du compresseur basse pression, revêtement qui, à l'état de poudre, s'enflammait de manière explosive dans le compresseur haute pression en causant sa rupture générale le long de son plan de joint.

Ce genre d'incident s'était en effet produit au décollage en deux occasions et ses conséquences spectaculaires sur la sécurité de l'avion ont déclenché de manière non moins spectaculaire le remplacement très rapide du revêtement précédent des carters du compresseur basse pression par un revêtement non combustible en nid d'abeilles aluminium sur tous les réacteurs de ce type en service.

Ce mode d'incident illustre par ailleurs le rôle en interface des structures de réacteur sur leur intégrité structurale propre.

(2.3) Roulements principaux

La fiabilité mécanique et les conditions de fonctionnement des roulements principaux constituent sur le plan de la technologie, un facteur très important contrôlant l'intégrité structurale des réacteurs à taux de dilution élevé.

Il a été constaté par exemple, assez récemment, que l'écaillage progressif des billes ou du chemin intérieur du roulement à billes n° 2 d'un réacteur de ce type (figure 7) a pu entraîner un grippage complet de ce roulement et sa destruction ainsi que la rupture pratiquement complète des ailettes stator et rotor du compresseur haute pression dont ce roulement supportait la butée axiale.

Le développement de tels incidents peut être susceptible dans un cas particulièrement sévère de ne pas être totalement contenu, outre leurs conséquences économiques résultant des dépenses pour la réparation du réacteur intéressé de l'ordre d'un demi million de dollars.

On peut estimer à cet égard, que l'évaluation de la probabilité de vie d'un roulement principal sur des réacteurs de ce type en utilisant la limite dite "BIO" sur un diagramme de Weibull, c'est-à-dire la limite de vie assurant une probabilité de rupture de 10 % d'une population de roulements pour un mode de rupture déterminé qui était acceptable sur les réacteurs des générations antérieures comptant deux roulements de butée adjacents par rotor, est devenue inapropriée sur les réacteurs modernes dont la vie devrait semble-t-il être exprimée par un coefficient beaucoup plus restrictif tel que B-5 ou même B-2.

(2.4) Ingestion

Le problème de l'intégrité structurale des réacteurs modernes et en définitive de la sécurité du vol en cas d'ingestion d'objets étrangers et en particulier d'oiseaux au décollage ou à l'atterrissage a pris une dimension croissante sur les réacteurs à taux de dilution très élevé et à ce titre mérite une discussion particulière.

Outre que ce risque bien connu est aggravé par les conditions d'environnement sur certains aéroports favorables à la présence d'oiseaux souvent de grandes dimensions, on peut estimer à partir de l'analyse de nombreux incidents que les réacteurs précédents sont spécialement vulnérables aux ingestions et à leur conséquence pour deux raisons fondamentales parmi les divers facteurs du tableau 2 :

(a) Les taux de dilution de 4 et au-dessus se traduisent par une augmentation de débit d'air du fan au décollage telle que la dépression devant celui-ci permet de moins en moins aux oiseaux même de grandes dimensions d'échapper au débit d'air du fan. La zone de probabilité d'ingestion augmente en conséquence autour du réacteur.

(b) L'analyse d'impacts d'oiseaux sur les ailettes de fan et de leur trajectoires d'ingestion, montre clairement que la fracture locale d'une ailette entraîne la libération d'un fragment métallique centrifugé à l'extérieur du fan mais qui ne peut être absorbé à son tour au travers de celui-ci tant qu'il n'a pas été réduit à l'état de fragments de plus petites dimensions ce qui suppose que le fragment initial demeure un temps prolongé devant le fan en y aggravant progressivement les dégâts et les balourds secondaires (figure 8). Cet effet résulte de l'angle d'attaque très faible des extrémités d'ailettes de fan qui accroissent l'effet de masque géométrique s'opposant à la pénétration d'un objet extérieur suivant une trajectoire oblique.

Une expérience sur laquelle les statistiques abondent permet d'estimer que les règles actuelles de certification pour démontrer la capacité et la résistance aux absorptions d'oiseaux des réacteurs à taux de dilution élevé méritent d'être reconsidérées pour les raisons précédentes.

Il est en effet assez remarquable que les dégâts secondaires d'ingestion soient dans beaucoup de cas, par leur nature et leurs conséquences, sensiblement plus sérieux que ceux constatés après absorption d'oiseaux conformes aux spécifications et aux règles présentes de certification.

Compte tenu de ce qui précède, on peut par suite considérer désirable que la certification des réacteurs futurs dans ce domaine démontre l'absorption non de corps étrangers tels que des oiseaux mais aussi celle d'objets métalliques de même nature que le matériau des ailettes de fan, de masse et de dimension analogues aux fragments statistiquement susceptibles de résulter de fractures initiales d'ailettes par premiers impacts.

La perspective de l'utilisation croissante d'avions moyens ou courts courriers dans les années à venir équipés de réacteurs à taux de dilution élevé donne au problème des ingestions un intérêt accru surtout si l'on considère l'emploi de plates-formes d'ailettes de fan d'une configuration nouvelle.

(3) CONCLUSIONS

(3.1) On peut souhaiter que les remarques précédentes contribuent à souligner les domaines où l'expérience antérieure des réacteurs à taux de dilution élevé justifie des améliorations en matière de résistance et d'intégrité structurales c'est-à-dire de sécurité de fonctionnement et de sécurité de vol.

(3.2) Si l'on devait donner une priorité aux objectifs les plus importants, il apparaît que les orientations devraient faire l'objet en premier lieu de programmes dans les domaines du titane, des ingestions, des ailettes compresseur et turbine, du carburant et du lubrifiant, et qu'en définitive, le rapport poussée/poids vu sous cet angle est probablement un des paramètres les plus influents et les plus importants à considérer.

(3.3) L'expérience montre que le développement du réacteur une fois en service ne peut permettre que des améliorations très limitées pour réduire des risques dont il importe de prendre pleinement conscience au stade de la conception initiale.

(3.4) Un bien meilleur degré de compromis du rapport poussée/poids que celui pratiqué jusqu'à présent s'avère donc nécessaire en acceptant certaines augmentations de poids judicieuses qui pratiquement sont impossibles à incorporer plus tard pour des raisons techniques ou en raison de leurs coûts exagérés pour les utilisateurs.

La tendance néanmoins à l'augmentation de poids, toutes raisons comprises, une fois en service justifie la position précédente. La figure 9 montre comment cette augmentation évolue sur un réacteur récent à taux de dilution élevé au cours des cinq premières années après sa mise en service initiale. Plus d'une centaine de modifications représentant plus de 100 kgs ont contribué à cette évolution au prix de difficultés innombrables et de dépenses de tous ordres extrêmement importantes pour les utilisateurs. Encore faut-il souligner que malgré cela, les modifications résolvant certains des risques structuraux discutés plus haut, restent à étudier et à appliquer et, pour le moins, sont problématiques.

(3.5) La politique d'entretien, rappelée dans le tableau 2, ne peut elle seule malheureusement se substituer d'une manière satisfaisante aux améliorations qui peuvent paraître nécessaires. Tout au plus et au prix de contraintes non négligeables, la mise en oeuvre de moyens de surveillance d'état sur avion permet-elle de limiter très partiellement la probabilité et les coûts des risques structuraux par leur détection avancée dans certains cas et pour des périodes limitées.

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NATURE DES RISQUES STRUCTURAUX	
Thermiques	Feu de carburant Feu de lubrifiant Fusion partielle d'éléments structuraux statiques Feu de titane
Cinétiques	Ruptures d'ailettes de compresseurs ou de turbines Perforation d'éléments structuraux statiques Rupture de disques Rupture de roulements principaux Non containement en général de fragments balistiques

Tableau_1

EFFET POTENTIEL D'UNE VARIATION DES CARACTERISTIQUES SUR LA SEVERITE DES RISQUES modéré + réduit – important + + sans changement 0		ORIGINE DES RISQUES STRUCTURAUX							
		THERMIQUE			CINETIQUE				
					Origine Réacteur				Origine Extérieure
CARACTERISTIQUES REACTEUR		Carburant	Lubrifiant	Titane	Disques	Ailettes	Roulements	Structures	Ingestion
Performances	Poussée de décollage	+ +	0	+ +	+	+ +	+	+	+ +
	Rapport Poussée/Poids	+ +	0	+ +	+ +	+ +	+	+	+ +
Cycle thermo dynamique	Débit d'air	0	+	+ +	0	+	+	0	+ +
	Taux de dilution	0	0	0	0	0	+	0	+ +
	Taux de compression	0	+ +	+ +	0	+ +	+	+	+ +
	Rendement du cycle	+ +	0	+ +	0	+ +	0	0	0
Conception basique	Contraintes		0	+	+ +	+ +	+	+	+ +
	Gradients de température	+ +	0	+ +	+	+	0	+	0
	Limites de vie	0	0	+	0	0	0	0	0
Technologie	Jeux réduits en bout ailettes compresseur	0	0	+ +	+	+ +	+ +	+ +	+ +
	Marges de pompage réduites	0	0	+	0	+	0	+	0
	Nouveaux matériaux	0	0	+	+ +	+ +	0	+ +	+ +
	Joints d'air de compresseur	0	+ +	0	0	0	0	0	0
Politique d'entretien en service	Modulaire	0	0	0	0	0	+	0	0
	Selon état	0	0	0	0	+	0	0	0
	Techniques de surveillance	–	–	–	0	–	–	0	–

Tableau..2 Analyse de l'importance et de l'origine des principaux risques structuraux

NATURE DES RISQUES STRUCTURAUX	
Thermiques	Feu de carburant Feu de lubrifiant Fusion partielle d'éléments structuraux statiques Feu de titane
Cinétiques	Ruptures d'ailettes de compresseurs ou de turbines Perforation d'éléments structuraux statiques Rupture de disques Rupture de roulements principaux Non containement en général de fragments balistiques

Tableau_1

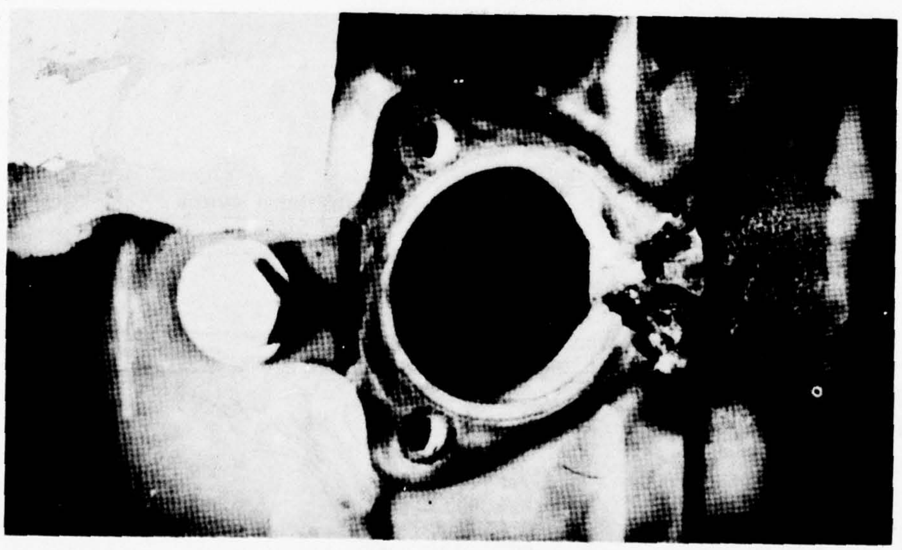
FIG.1

Fusion et perforations locales carter diffuseur par fuite de carburant

FIG.2

Fuite d'injecteur de carburant

FIG.3



Perforation et fusion locales carter diffuseur par fuite de carburant

Réduction des Gradients de Température sur un Injecteur de Carburant

FIG.4

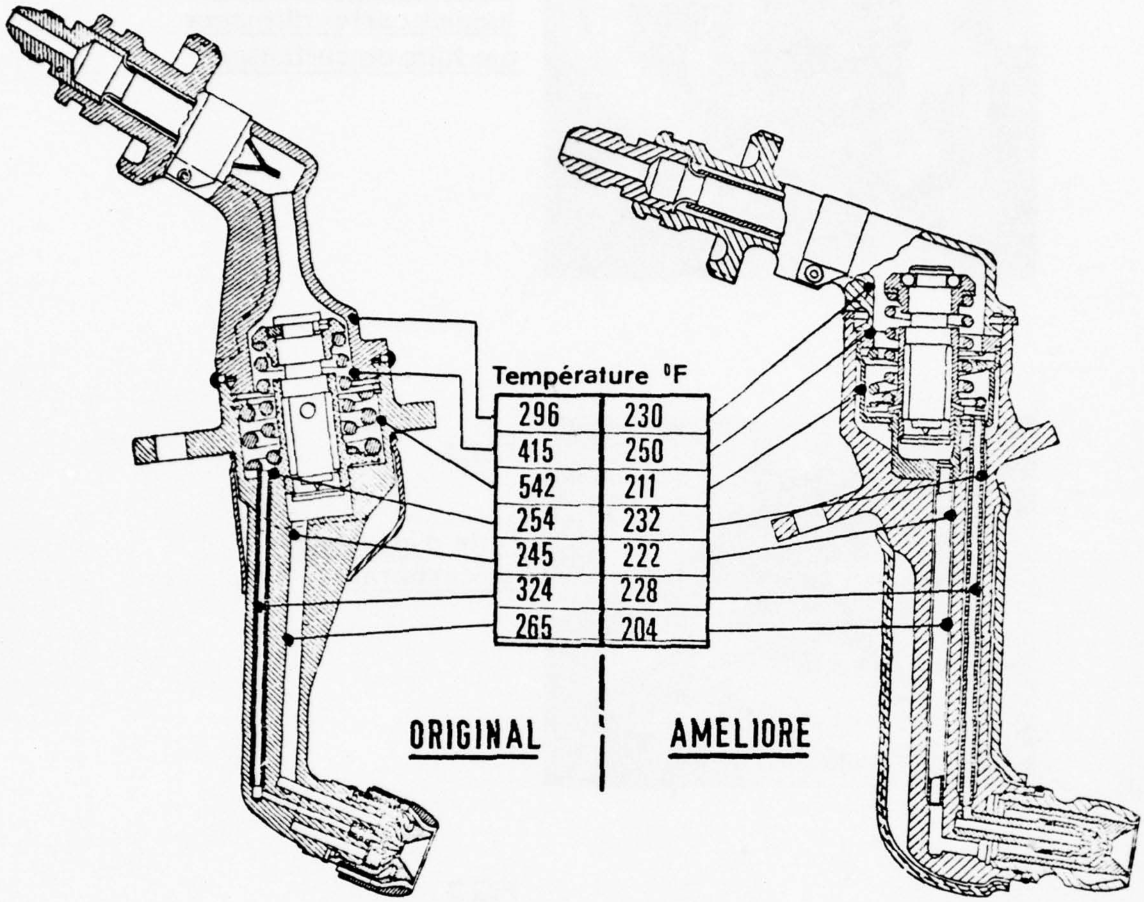


FIG.5

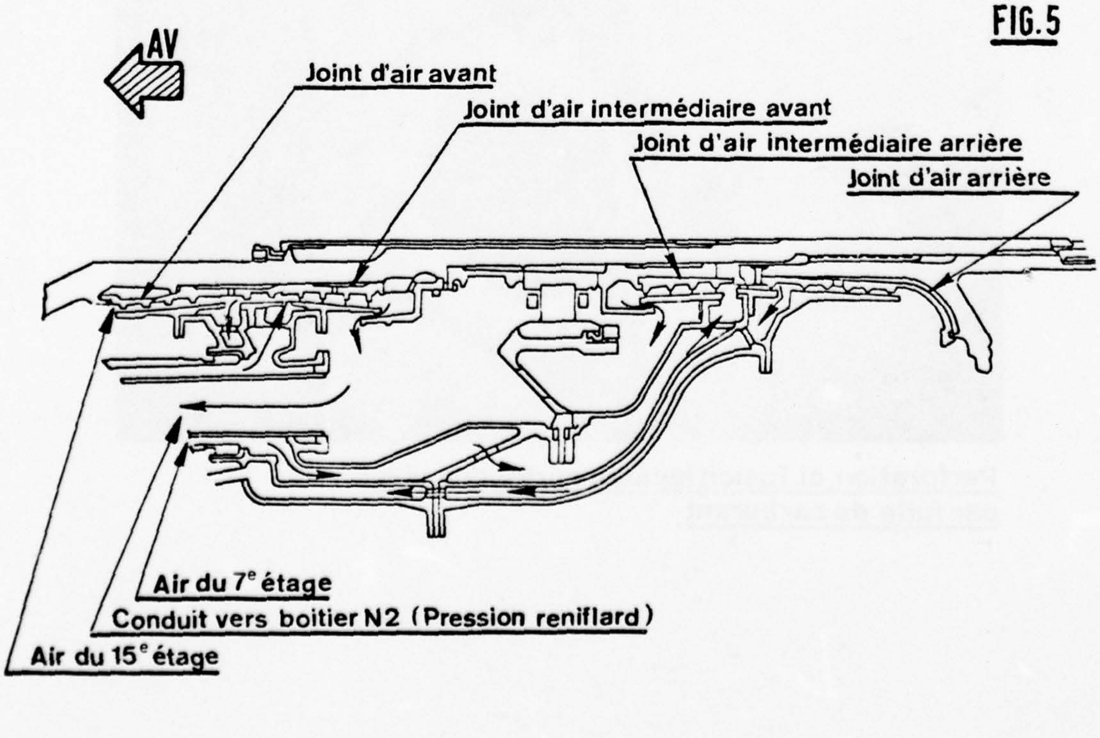
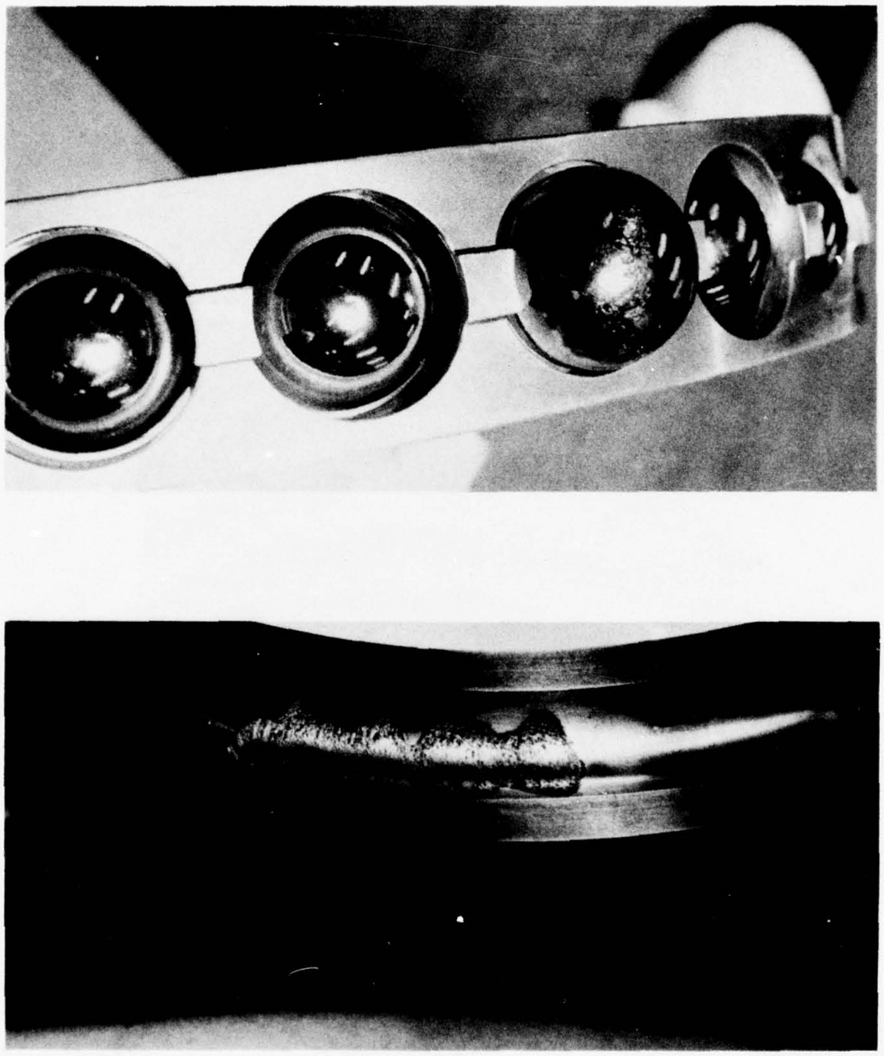


FIG.6

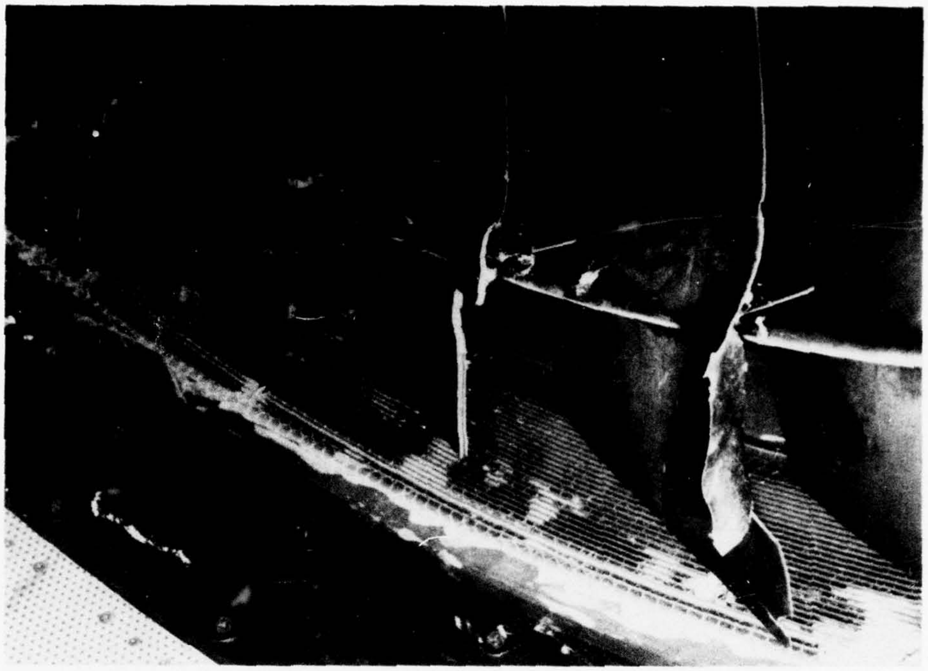
Effet d'un feu de titane sur le carter compresseur haute pression

FIG.7



Ecaillage de roulement principal

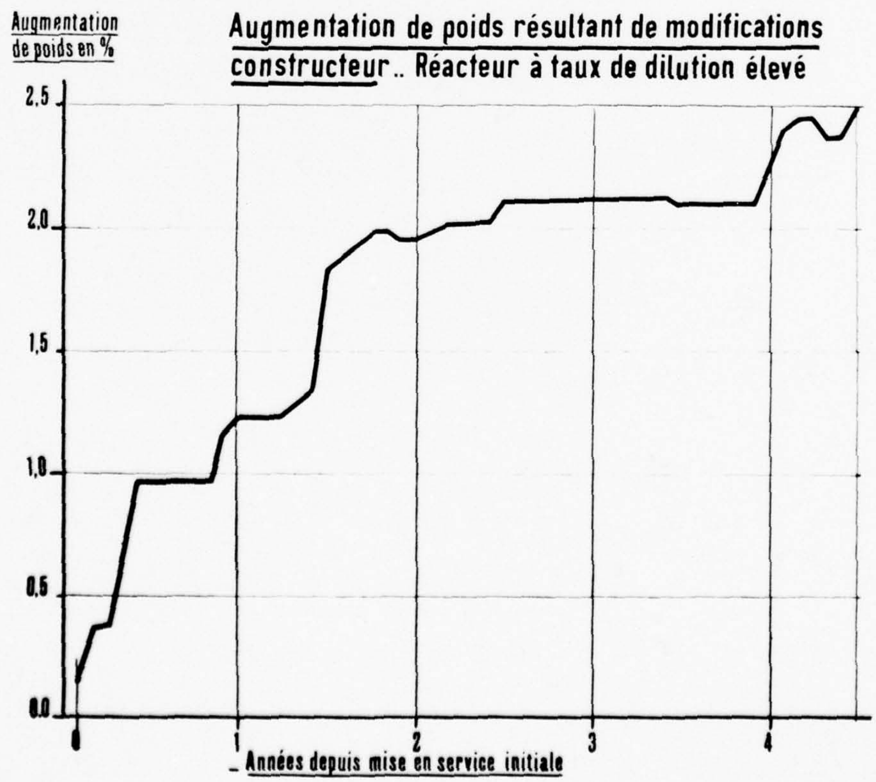
FIG.8



Ruptures locales ailettes de fan par ingestion d'oiseaux

FIG_9

Augmentation de poids résultant de modifications constructeur.. Réacteur à taux de dilution élevé



DISCUSSION

J.Slatford

May I make a comment on titanium fires? We already have a regulation controlling the unlimited use of titanium, and at the moment we are in the process of enlarging that and making it more specific.

Secondly, a comment on bird ingestion: I must talk here on the RB 211 which is the engine I know best. That has met our current requirement and in fact the service record I think is incredibly good. I think there has been one in-flight shut down in something like two million hours. And our real problem has been with the small fan jet or jet engines where we can scale down the engine size but have found no way of scaling down bird size. And this is a real problem.

Réponse d'auteur

On peut néanmoins souligner deux points; le premier c'est que dans cette présentation il n'a pas été sous-entendu que les moteurs à grand taux de by-pass actuellement en service donnaient tous lieu, de manière identique, aux même genre de problèmes, il est certain que certains sont plus sensibles ou sont plus critiques que d'autres et si vous voulez le but est néanmoins d'essayer de souligner que certains progrès peuvent dans certains cas être encore possibles, en tout cas désirables, d'après l'expérience que nous ayons eue.

En ce qui concerne l'emploi du titane, nous savons effectivement que certaines régulations existent actuellement dans son emploi, mais néanmoins, il apparaît quand on regarde les choses de très près la mise en oeuvre du titane, qui par ailleurs est un matériau extrêmement important, devra probablement s'accompagner de précautions supplémentaires relatives, résultant des caractéristiques de combusion du titane et des conditions dans lesquels il est mis en oeuvre et probablement certains progrès sont encore très possibles, compte tenu d'expériences très récentes.

DEVELOPMENT PROCEDURES TO PROMOTE RELIABILITY

by

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Great Britain

INTRODUCTION

1 This Paper concentrates on reliability attainment so far as the military aircraft gas turbine is concerned. Early reference is made to civil engine development and operational reliability in order to emphasise that the aircraft gas turbine particularly in a basically non-complex form did provide a step change in reliability when compared with its predecessor, the high-powered (2,000-4,000 HP) piston propeller engine. Sir Frank Whittle's genius lay not only in the fact that he pioneered the developed application of the gas turbine to jet propelled flight, but that he recognised that an essential requirement was simple sound engineering within, what is called today "the state of the art". He chose to employ the double-entry centrifugal compressor directly driven by a single stage turbine to reduce cross-sectional area of the power plant and used a contraflow multi-chamber combustion system to provide a short shaft (and thereby minimise whirling problems) and to shield the turbine blading from direct flame radiation from the primary combustion zone. The fact that the reverse flow combustion arrangement was restrictive when it came to increasing the turbine flow capacity to achieve higher performance by up-speeding and raising the temperature was accepted for the first generation engine. The obvious move to a "straight through" combustor introduced the first step towards reduced reliability stemming from what was to become an insatiable quest for higher and higher performance. However, the problems associated with the increased shaft length were soon overcome, although expensively as they did not show up until the engines had been in operational service for some while. In the UK, the Derwent, Goblin, Nene and Ghost centrifugal engines can be fairly described as having a good reliability, whilst in the USA the successful I-series of centrifugal engines modelled after the Whittle jet engine, and the licence-built P&W J-42 and J-48 engines achieved similar reputations. The ultimate reliable aircraft centrifugal gas turbine is perhaps the Rolls Royce Dart, a nominal 1,000 SHP turboprop. In spite of the complexity of a change to a two-stage compressor and multi-stage turbines (later cooled), and a progressive erosion of potential reliability by up-rating to 2,000 + SHP, the engine has maintained its very high reliability record.

2 The application of the axial flow compressor to the military area was inevitable because of its smaller frontal area and the promise of a maintained high efficiency at pressure ratios beyond 6:1. However, the relative fragility of the compressor system which can be equated to a myriad of thin cantilevers fighting desperately to survive in a hostile environment, coupled with the inevitable increase in length of the engine introduced design complexity outside the bounds of experience particularly with weight at a premium. The German axial flow engines of WWII (Jumo 004 and BMW 003) were very unreliable not because of any lack of design competence (in fact notable performance levels were attained) but because the non-availability of strategic materials made necessary the complexities of a variable final nozzle, significant compressor blow-off

and a sophisticated all-speed governing system as well as a complex air cooling arrangement. Even today, the new generation of engine designers could learn a lot from careful study of those engines. Both in the USA and in the UK, innovative axial flow engine design benefitted from the experience of companies with related experience in the non-aviation field, notably Metropolitan-Vickers in Britain and the General Electric Co in the USA. With the benefit of the pioneering effort, the established aircraft engine industry developed, at significant cost to the taxpayer, the axial flow engine to higher performance levels to a degree where it virtually ousted the centrifugal engine in military combat aircraft. Pratt and Whitney, Rolls Royce and the Bristol Engine Co were the prime manufacturers who successfully developed the two-shaft higher pressure ratio axial flow turbojets and turbofans. The General Electric Co elected to develop the multi-stage variable compressor stator single-shaft engine, their J-79, which has successfully powered a wide range of military aircraft including the Phantom, operational world wide. Thus is seen the increasing complexity of the engine as performance goals were set higher and higher. As engine complexity increased, thereby absorbing the full effort and attention of the development engineers, reliability was left to be improved after the engine had been formally qualified. That the two-shaft axial flow turbofan can attain a very high degree of reliability is shown by the distinguished record of P and W's civil JT3-D engine. In this particular case, the core of the engine was very well developed by the military as the J-57. Having been in service for many years it has set a standard for reliability that will be hard to beat.

3 It is only recently, during the past 5 years or so that in the military arena Cost has become a significant factor. The relative decline in importance of the military combat aircraft which now competes for funding against equally or even more costly tactical and strategic weapon systems has had a profound effect in recent years.

4 It can be argued that the over abundance of defence funding which more than compensated for the high cost of increasing engine complexity also encouraged a persistence of development and qualification practices more appropriate to piston engines. The 150 Hours' Type or Model Test was the only real milestone which had to be passed and rather too high a proportion of applied effort has always been devoted to dealing with the specific nuances of the Type Test. At this stage, it is relevant to quote from an authoritative address (Ref 1) given in March 1953

> "To be first in producing good engines it is essential that you are enthusiastic; and then you must create an organization that can make break and remake and, finally, produce engines more quickly than your competitors. The ability to 'make and break' engines rapidly, and to keep a flow of experimental engines that practically swamps the test benches, is the only known way of successfully developing an aviation engine".

This philosophy of "run, break, re-make and run again" which leads to extravagent and ill-planned development has been slow to disappear and has been responsible for an over-abundance of very expensive scrap components at the end of many engine development programs. A proper understanding of the very complex process of engine development is given in Ref 2 which was presented in the UK in 1971. In this classic Paper, Mr E C Simpson rightly exposes and endorses the exploratory development/demonstrator approach to engine development so essential for ultimate success.

5 The preceding paragraphs are intended to explain how, in the past, development has concentrated primarily on achieving increased performance with little thought about consequences of the attendant considerable increase in complexity and cost. It has now been found through hard experience that reliability, along with cost and complexity, has generated problems that have eroded combat readiness.

6 Since the early 1960's, both US Military and UK general engine specifications (MIL-E-5007, MIL-E-8394, and D Eng D 2100 and 2300) have been extended to include tests supplementary to the 150 Hours' Military Qualification Test. Fig 6 extracted from Ref 1 compares the 150 Hour Type Approval Tests as they were in 1953. It is fair to state that the US test was then relatively more demanding and perhaps more realistic particularly so far as the number of starts and accelerations are concerned. Continuing collaboration has resulted in the two countries' Specifications remaining "in step" although the UK lower limit of temperature ie -26°C departs from the more realistic -40°C of the American specifications which also applies in the Civil Aviation area.

7 In the USA, a most determined effort has been made to properly qualify the P and W F100/F401 engine ahead of the engine's entry into service in the F-15 and F-16 aircraft. The work carried out is summarised in Ref 4. The testing subjected the engine to the most severe conditions expected to be encountered in its service life. Changes were made to the engine ahead of the formal 150 hour Test which was performed on an engine configuration to production standards. It is claimed that the testing provided early development action rather than reaction to flight problems. An indication of the duration and intensity of the testing is indicated by 1,850 hours of fan rig testing, 3,000 hours of Government Laboratory (AEDC and NAPTC) altitude testing, and 92 hours at Maximum Cycle Temperature at pressure levels above atmospheric up to 34 psia and inlet temperatures up to 445°K. Whether this very high level of pre-service entry development has paid off remains to be seen.

ENGINE USAGE IN SERVICE

"When you can measure what you are speaking about and express it in numbers you know something about it, but when you cannot your knowledge is of a meagre and unsatisfactory kind" Lord Kelvin (1824-1907)

8 An examination of the historical record will reveal a glaring gap in the military engine development process in that little or no work has been done to obtain quantified accurate information on the actual way in which engines are used in military service. It is only now that, belatedly, the situation is being rectified. "Mission profile analysis", "Engine usage spectrum analysis" are now commonly heard phrases. It is interesting to reflect that military Engine Condition (Health) Monitoring programs have been carried out without apparently sufficient realisation that the condition of an engine is very much dictated by the way in which it has been used (or abused!). Over many years the hapless engine designer having seen the margins embodied in the engine by the use of well-judged design factors and the proper selection of materials eroded away by the insatiable requirement for higher performance, has not been provided with a feed-back of reliable and quantified data for him to re-design for reliability. This situation has now been realised initially because of the safety, logistic and cost implications of low cycle fatigue problems with primary rotating engine components eg discs and shafts. Work is now in hand in the UK to provide a data base from which to generate design criteria and which also will be used to change development procedures to promote reliability.

9 When the work was initiated some 2-3 years ago, it was found that there was no "off the shelf" up-to-date, cost-effective system for airborne data acquisition and subsequent ground analysis. The first flight program on two RAF Harriers was carried out using a UV (ultra-violet) trace recorder. The Engine Directorate of UK MOD(PE) initiated the design and development of EUMS ("Engine Usage Monitoring System") based on digital data acquisition and formatting with subsequent data analysis followed by print-out and, recently a parameter analogue display. The System evolution and description are presented in Ref 3. The System has performed well from the on-set and is being expanded in terms of both capability and capacity to meet a variety of requirements.

During the initial development of the System, the Service-imposed limitation of only recording engine rpm excursions was not popular as from the onset the equipment had been designed to accommodate a larger range of parameters. In fact it proved to be a blessing in disguise as the development problems with the DAU (Data Acquisition Unit) and of the QAR (Quick Access Recorder) were identified more quickly and positively without attendant problems with unreliable sensors, or connectors and wiring. Figs 1 to 5 have been extracted from Ref 3 and show the System as it is in use today.

10 Already some very valuable and unique results have been obtained and changes in lifing of engines are being clearly indicated. The ability to define in more detail the Service usage is a step change in quantifying lifing of engines but more importantly, changes in the overall philosophy in engine design, development testing, maintenance and reliability are being signalled.

11 The current programme is correlating low cycle fatigue consumption, based on rpm excursions (and in relevant cases a combination of rpm and temperatures) with flight hours. However it is important that the existing practice of relating LCF with hours flown is examined in depth. Fig 7 shows the variation which exists in a typical Service aircraft application in cycles per flight and number of flights of the same sorties. Fig 8 shows the variation of flight cycles related to different sorties flown. Dependant upon the percentage sortie flying by a fleet of aircraft, then an overall 'weighted' LCF usage value may be calculated. It is self evident from the two figures that a precise knowledge of engine usage enables more economic management to be made of life assuming that the logistics associated with this different approach are accepted. A continuing action within MOD is to optimise the use of this data to guide and refine the decisions taken by Lifing Technical Committees and more importantly to challenge the basic concepts of lifing management.

12 Further, the data which has been produced provides another immediate impact on assessing the effectiveness of existing test procedures. Much data has been generated over the years on test bed running leading progressively into recording of data during the flight development phase. Correlatory work has established read-across on performance aspects of the engine, and there it would seem the story might end. As highlighted earlier in the paper there is little or no resemblance between development and actual Service flying conditions. Equally, performance is achieved in Service, but at what expense?

13 The EUMS activity is already pointing the way on how to achieve a much more realistic development test programme for 'real' reliability. Furthermore it is a 'living' progressive tool which is able to quantify and dictate the format of testing to suit the changing needs of a flexible fighting unit. The 'loop' is closed.

14 The usage of an engine for a particular sortie will produce regions or patterns of rpm changes with corresponding temperature excursions. Fig 9 and Fig 10 indicates the way in which the data is used to compile meaningful test schedules. (The corollary with Fig 6 is self evident). Suitable weighting of these values to reflect the Service percentage flying of the different sorties will produce a test datum having a high level of confidence. Due regard to other environmental conditions of the engine ie OAT, Airspeed and Altitude will of course be necessary.

15 In relation to the Pegasus/Harrier EUMS recorded data (which includes six engine and three aircraft parameters), we are extracting the maximum amount of information on the engine usage profile and have initiated a system to correlate engine defects/strip condition and maintenance history with the EUMS data. Programmes additional to LCF include Creep and Thermal Usage are already being compiled. Data on engine history from the Pegasus Service record cards is already being compiled.

Looking to the future, we are extending both the versatility and the capacity of "EUMS" to become "EU/LMS" (Engine Usage/Life Monitoring System). The incorporation of a digital microprocessor and relevant peripheral electronics such as read-only and re-programmable memories plus further modification of the cassette tape transport will signficantly improve the equipment's capability. The addition of a VDU (Visual Display Unit) converts EUMS into a versatile and cost-effective Engine Health Monitoring system.

CONCLUSIONS

1 Virtual total concentration on the achievement of performance goals has resulted not only in increasingly complex and therefore very costly engines, but also in a near total lack of attention to design for reliability.

2 The resulting situation where reliability improvement is sought only after entry into Service,whilst providing interesting work for design and development teams, is now financially intolerable to an extent that operational readiness is seriously diminished.

3 Belatedly, the need for mission profile analysis programs to establish engine usage in service has been recognised and accepted by the Operators.

4 A continuous recording system (acronym EUMS) has been successfully developed and is being applied to a statistically viable number of various military engines in service. Starting with a prime target of establishing realistic LCF/ Flight Hours exchange rates for specific engine/aircraft combinations, thermal fatigue, creep and thermal shock factors (hot-section factors) and their synergism are now being investigated.

5 The EUM System provides a cost-effective base for an on-board EHM System. This is now made possible by the recent dramatic progress in electronics' LSI and the microprocessor

NOTE: LCF = Low Cycle Fatigue

EHM = Engine Health Monitoring

THE VIEWS EXPRESSED ARE THOSE OF THE AUTHOR AND DO NOT NECESSARILY REPRESENT THOSE OF THE UK MINISTRY OF DEFENCE (PE).

REFERENCE

1 "The Birth of An Engine British Practice in Aviation Engine Procurement and Development" An Address before the Institute of the Aeronautical Sciences, Cleveland, USA. 13 March 1953.

2 "Scar Tissue and Aircraft Propulsion Development" A Paper presented to the UK Royal Aeronautical Society by Mr E C Simpson, Director, Turbine Engine Division, AFAPL, WPAFB.

3 "The Development and Utilisation of an Engine Usage Monitoring System with particular reference to Low Cycle Fatigue Recording" by R Holl and R S Wilkins (UK, MOD(PE)). March 1976.

4 "F100/F401 Engine Development Testing Oriented to Real Life Conditions" by B J McDonnell and E W Creslein.

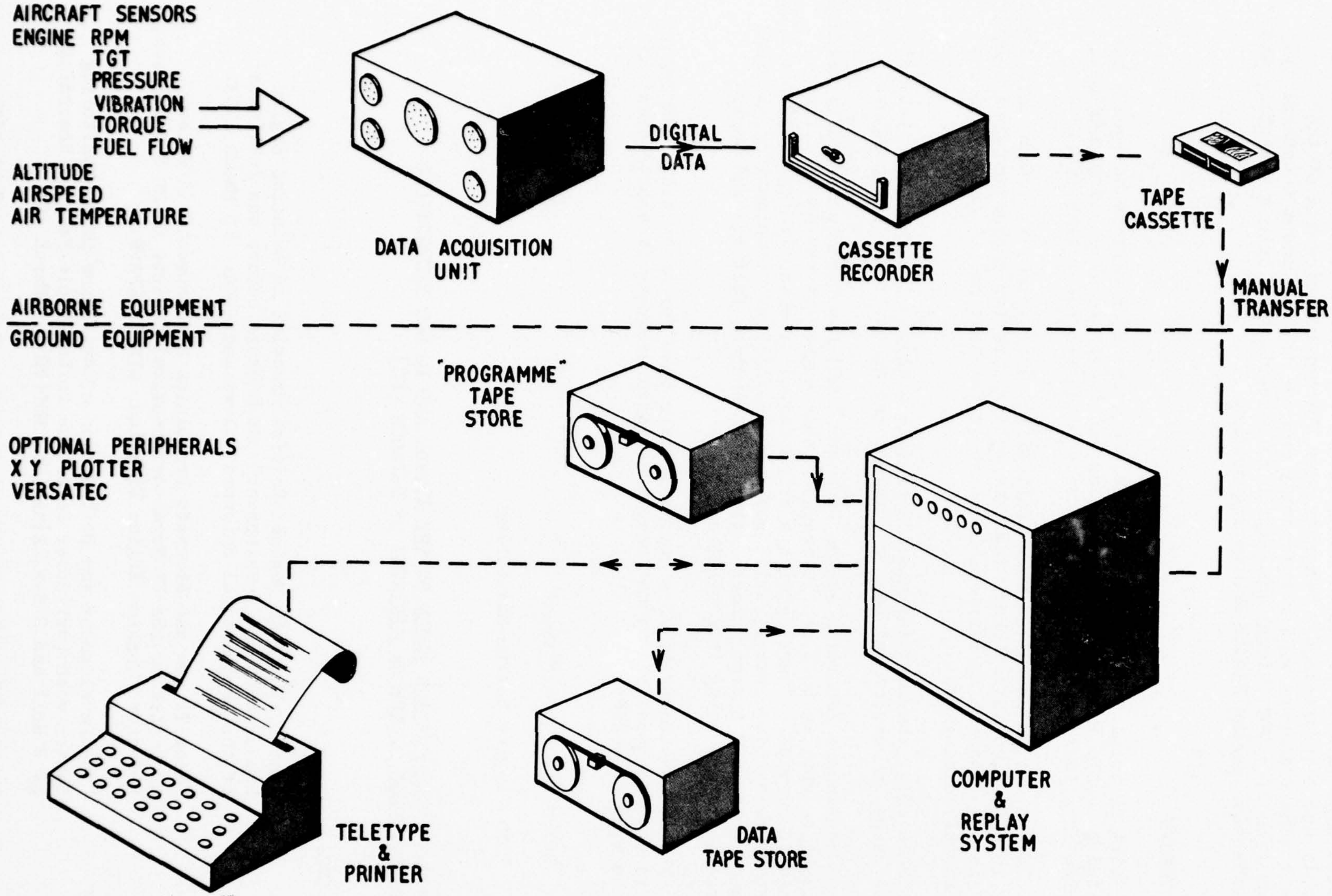


Fig.1 System diagram

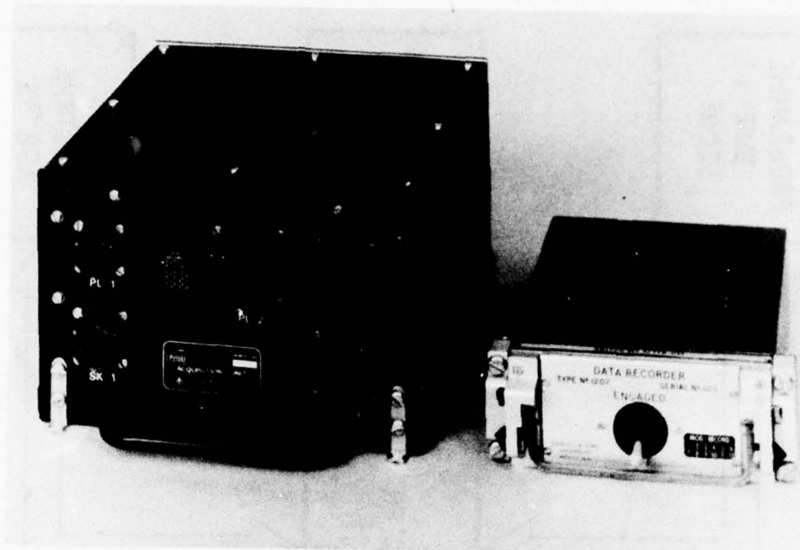


Fig.2 Airborne equipment

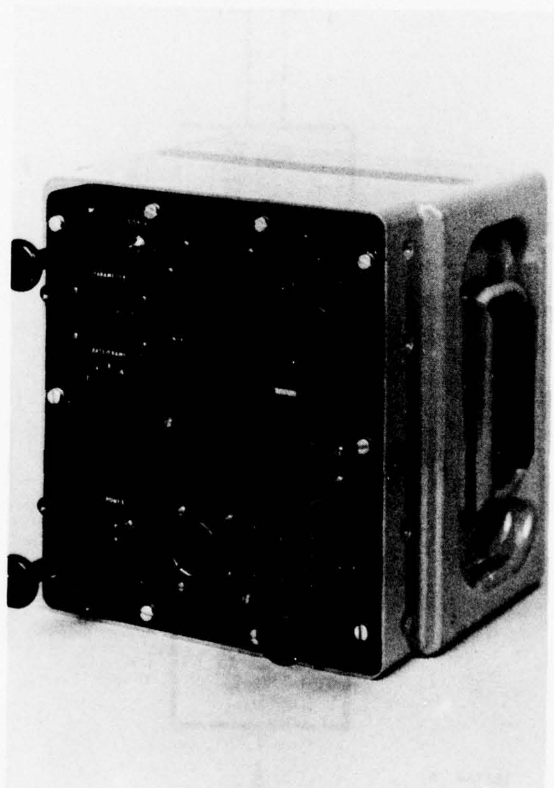


Fig.3 Flight line test set

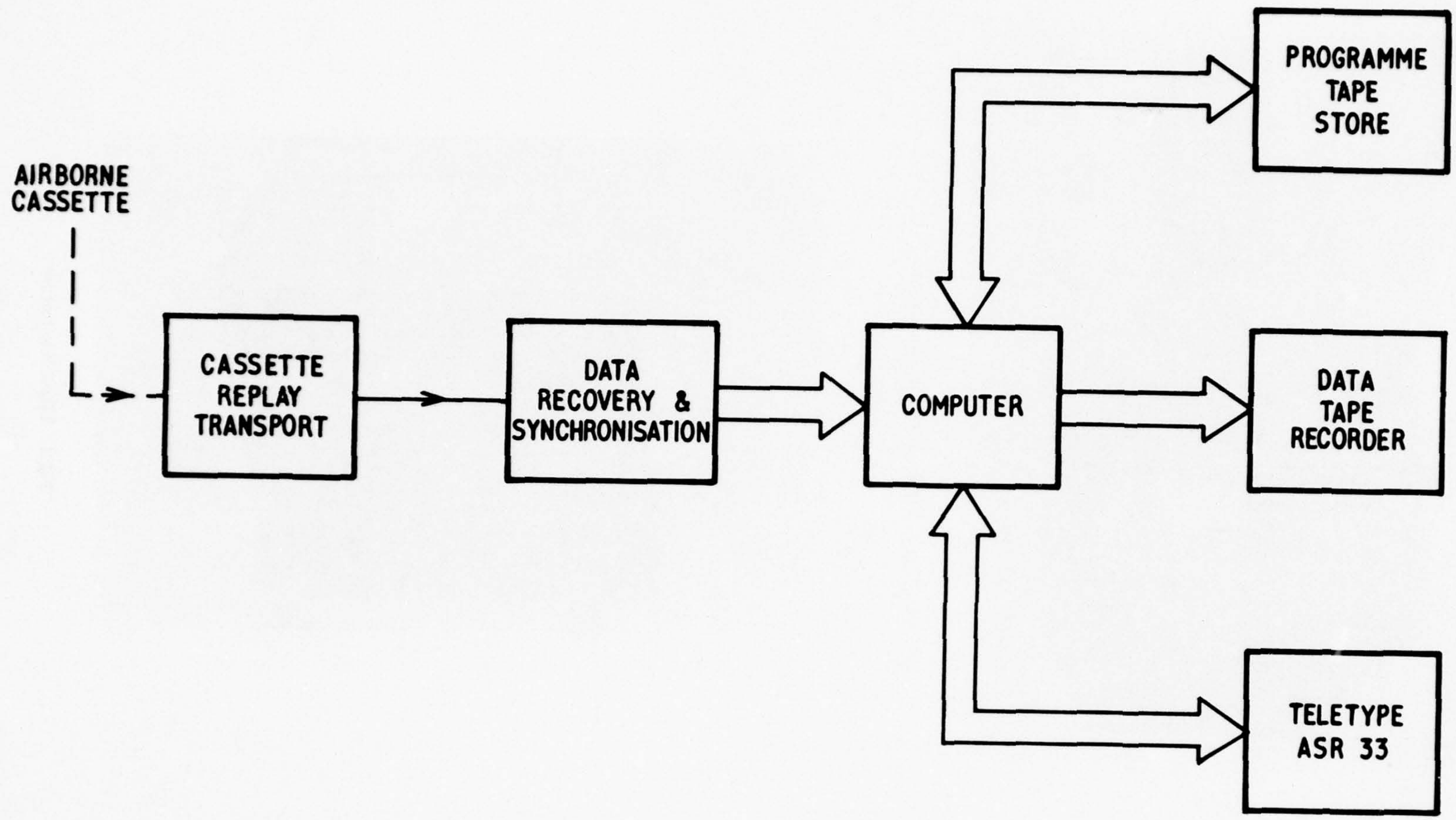


Fig.4 Block schematic – ground system

VULCAN OR PROVOST? (V OR P): P
TAPE UNIT? (Ø OR 1): Ø
L.S.D.(%): 1.5

AIRCRAFT TYPE/MK: JP5
AIRCRAFT REG: XW288
ENGINE NUMBER: 121Ø5
FLIGHT NUMBER: 36
DATE: 12/8/75 FUEL S.G:
TAKE-OFF TIME: 11.25
CALIBRATION NUMBER: 1
COMMENT: SOR L23

THIS FLIGHT (Y OR N): Y

ENGINE 1.

+43.Ø	+41.4	+63.Ø	+42.6	+48.4	+41.1	+62.7	+41.6	+54.9	+41.7
+1Ø1.4	+58.4	+1Ø2.Ø	+99.4	+1Ø1.4	+82.6	+1ØØ.9	+57.6	+84.Ø	+56.9
+89.8	+54.1	+1Ø1.7	+54.8	+97.8	+53.4	+55.5	+54.Ø	+1Ø1.4	+81.3
+87.5	+71.8	+76.3	+73.9	+84.8	+53.6	+95.9	+88.Ø	+1ØØ.9	+54.8
+59.5	+53.1	+98.9	+93.1	+98.Ø	+53.Ø	+77.7	+52.Ø	+93.4	+78.Ø
+81.8	+68.3	+88.6	+72.1	+75.0	+51.1	+91.9	+51,9	+84.8	+75.2
+85.2	+7Ø.1	+79.1	+52.8	+72.8	+5Ø.4	+1Ø2.Ø	+75.2	+84.4	+5Ø.8
+52.6	+5Ø.2	+1Ø1.7	+61.2	+102.Ø	+51.Ø	+1Ø2.6	+57.3	+8Ø.5	+56.5
+1Ø1.4	+52.8	+1Ø2.6	+1ØØ.Ø	+1Ø2.Ø	+83.4	+1Ø2.Ø	+52.6	+94.9	+91.6
+94.9	+88.9	+94.4	+73.6	+87.3	+83.Ø	+96.4	+52.3	+98.6	+96.2
+98.Ø	+83.4	+1Ø2.0	+54.9	+1Ø1.7	+8Ø.3	+91.4	+72.7	+93.9	+54.9
+84.Ø	+73.6	+1Ø1.4	+83.4	+1Ø1.4	+6Ø.7	+83.Ø	+79.2	+84.Ø	+82.2
+84.Ø	+77.8	+1Ø2.Ø	+95.1	+1Ø1.4	+57.1	+1Ø2.Ø	+59.3	+1ØØ.Ø	+89.8
+94.9	+56.9	+1Ø1.4	+55.6	+1Ø2.Ø	+6Ø.Ø	+1Ø2.Ø	+57.3	+94.4	+81.8
+1Ø1.7	+57.6	+97.5	+88.9	+97.5	+58.6	+86.9	+8Ø.7	+1Ø1.4	+54.4
+1Ø2.Ø	+55.6	+59.2	+57.5	+94.9	+57.1	+1Ø1.4	+56.Ø	+82.4	+78.9
+87.3	+85.4	+1Ø1.4	+45.8	+58.9	+42.2	+59.5	+41.9	+76.6	+44.5
+66.9	+42.9	+45.1	+42.3	+65.4	+46.1	+57.6	+42.2	+49.Ø	+46.6
+5Ø.1									

U.T.S. : 8Ø
1ØØ% STESS : 26
INDEX : 3.15
O/P LEVEL : Ø.Ø2
TIME OF FLIGHT (MINS.) : 75

R.P.M.		MIN.STRESS	MAX.STESS	ALPHA	GAMMA
+83.4	+98.6	+18.Ø75	+25.275	+Ø.35775	+Ø.Ø3924
+68.3	+88.6	+12.127	+8Ø.428	+Ø.37631	+Ø.Ø46Ø2
+54.4	+1Ø2.Ø	+7.7Ø2	+27.Ø68	+Ø.82419	+Ø.54386
+52.3	+1Ø2.Ø	+7.117	+27.Ø68	+Ø.84228	+Ø.58235
+52.8	+1Ø2.6	+7.246	+27.385	+Ø.85174	+Ø.6Ø321
+51.Ø	+1Ø2.Ø	+6.75Ø	+27.Ø68	+Ø.85347	+Ø.6Ø7Ø8
+5Ø.4	+1Ø2.Ø	+6.596	+27.Ø68	+Ø.85814	+Ø.6176Ø
+5Ø.2	+1Ø2.Ø	+6.558	+27.Ø68	+Ø.85928	+Ø.62Ø18
+Ø.Ø	+1Ø2.6	+Ø.ØØØ	+27.385	+1.Ø5328	+1.17766

CYCLES Ø TO 1ØØ : +16.6179Ø
CYCLES/HR : +13.2943Ø
FINISH

Fig.5 Typical LCF printout

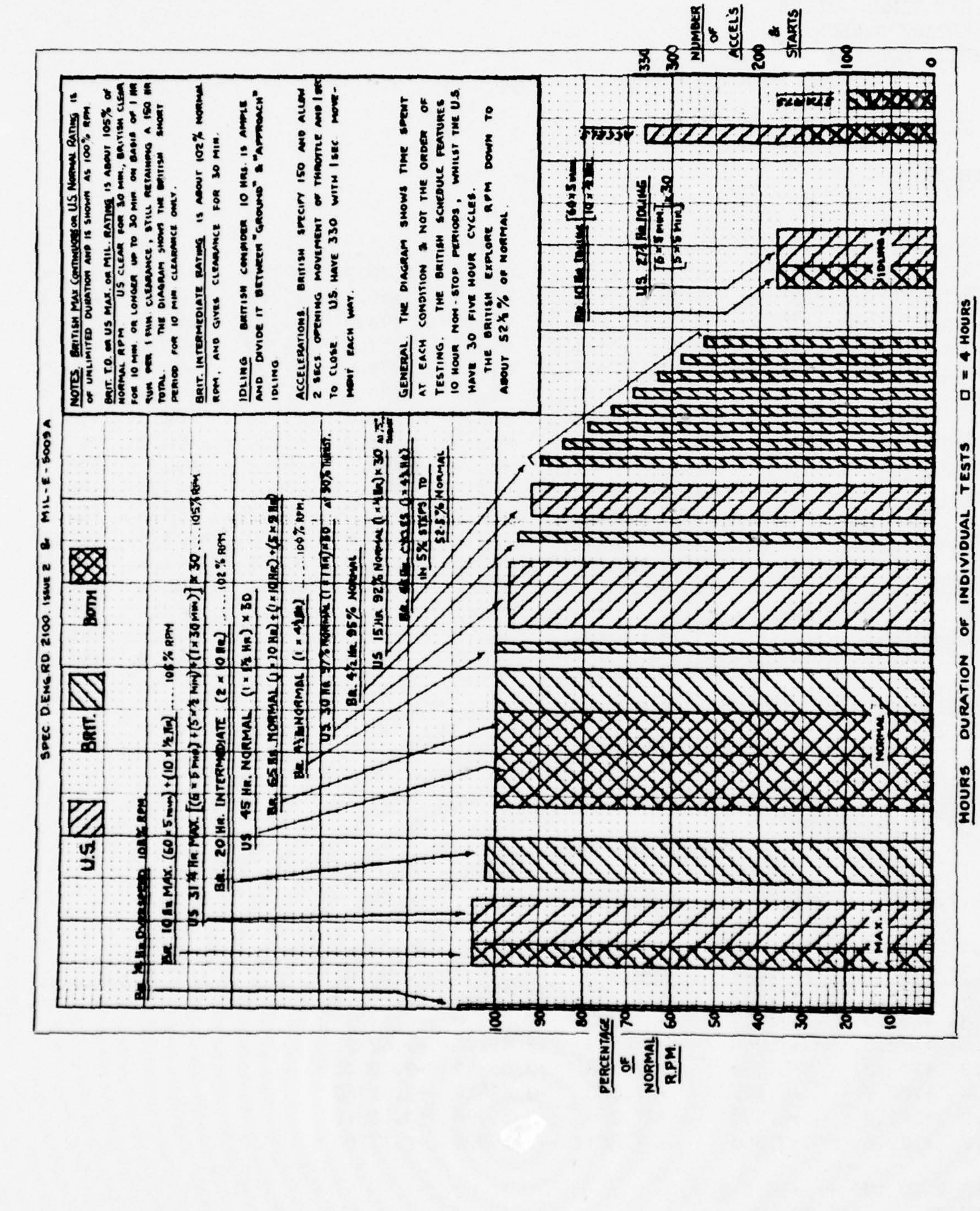


Fig.6 British and US turbine engine 150 hour type approval tests

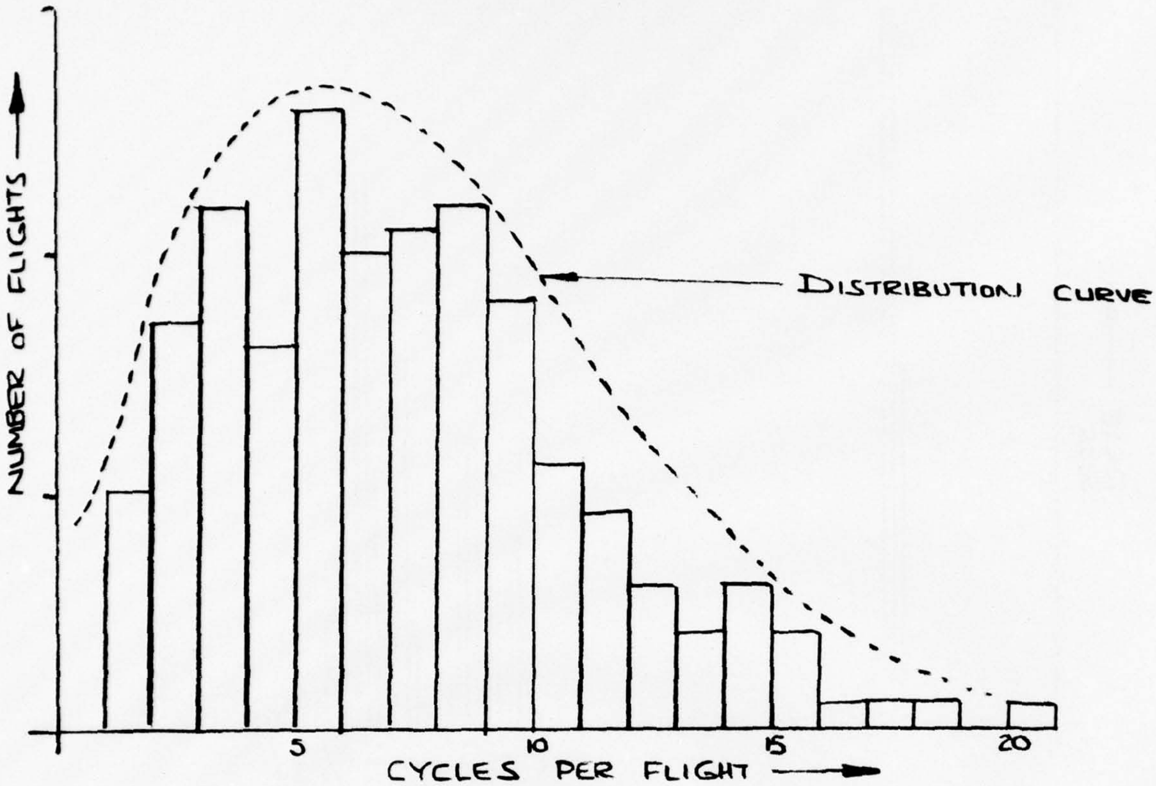


Fig.7 Variation in flight cycles/No of flights for one specific sortie code

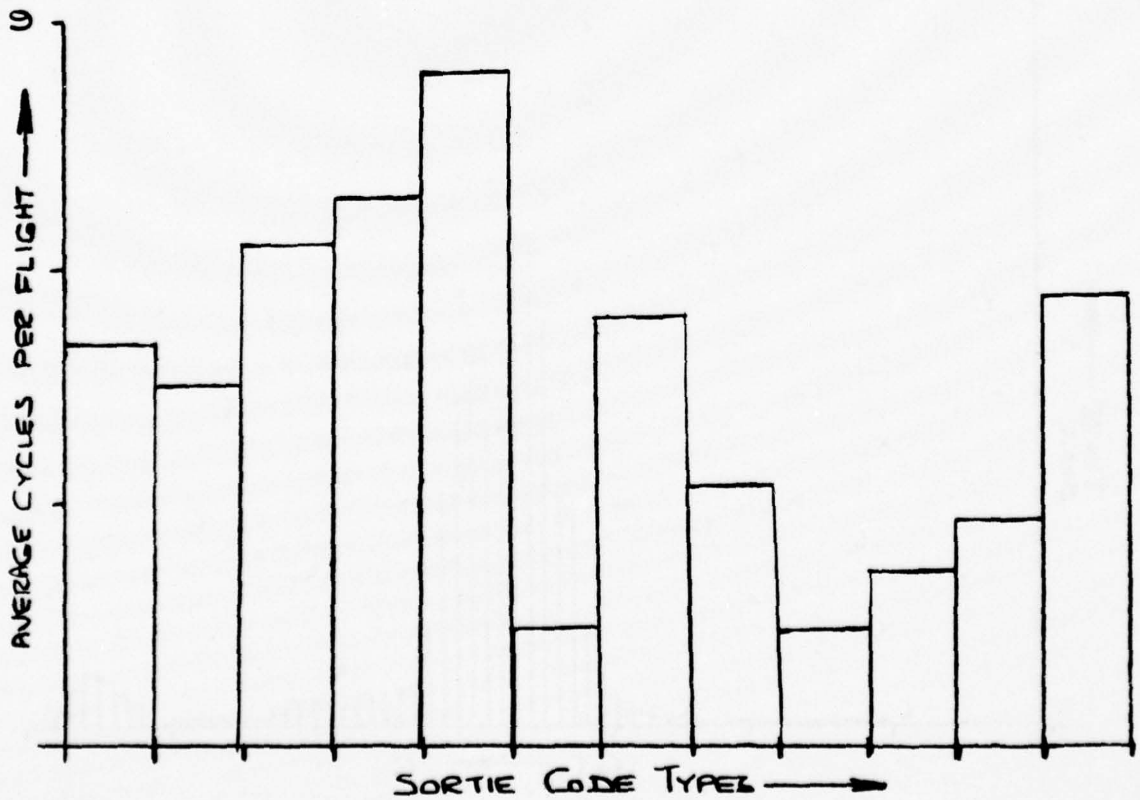


Fig.8 Variation in flight cycles/sortie code

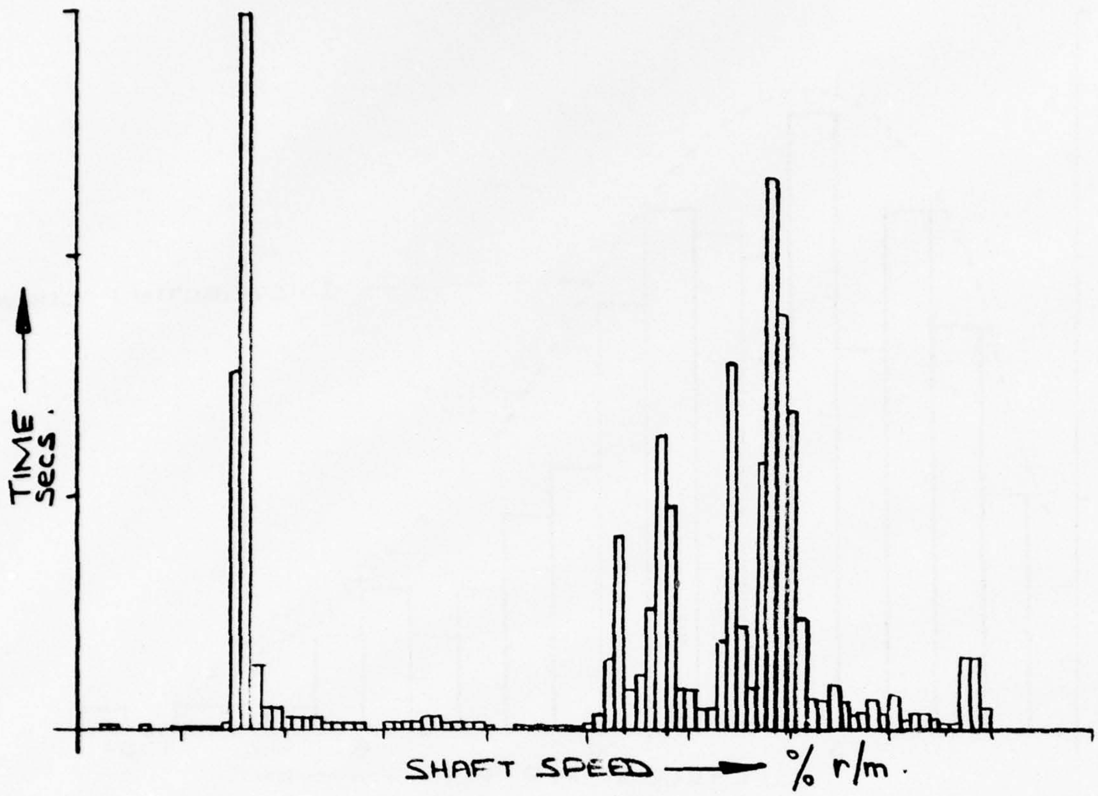


Fig.9 Typical in-service engine usage r/m pattern

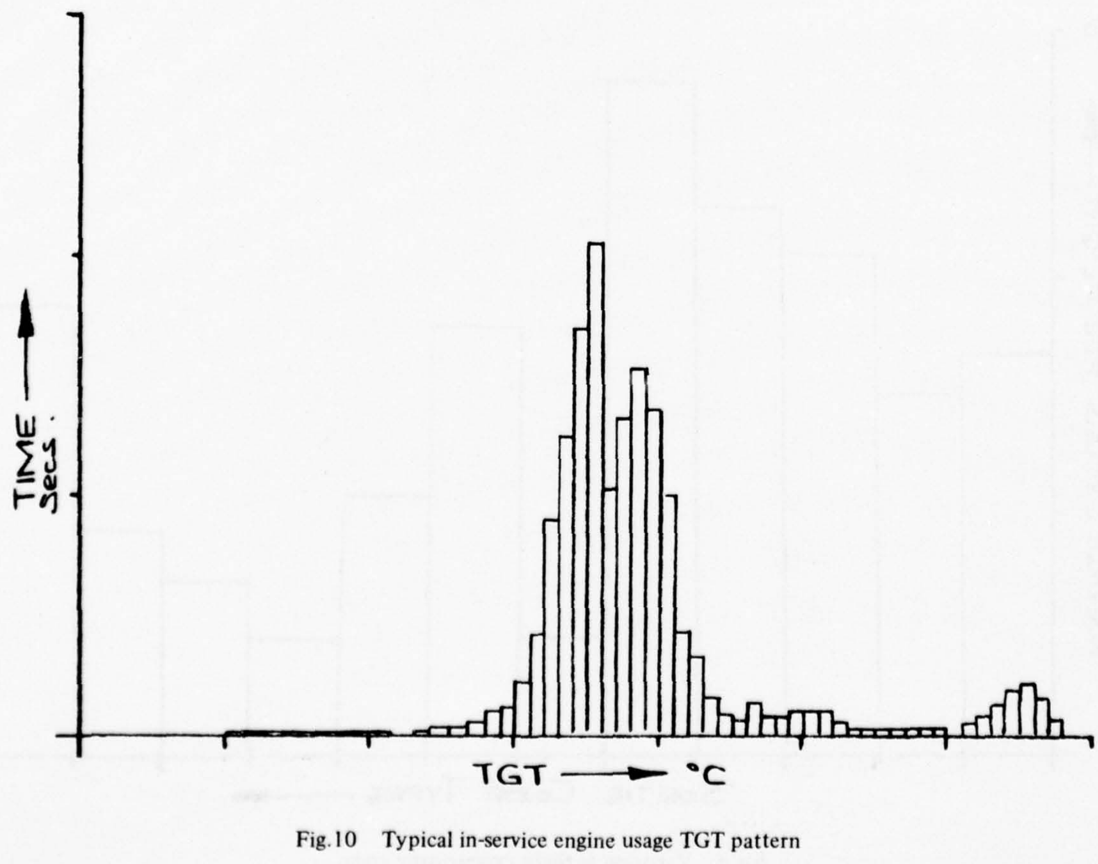
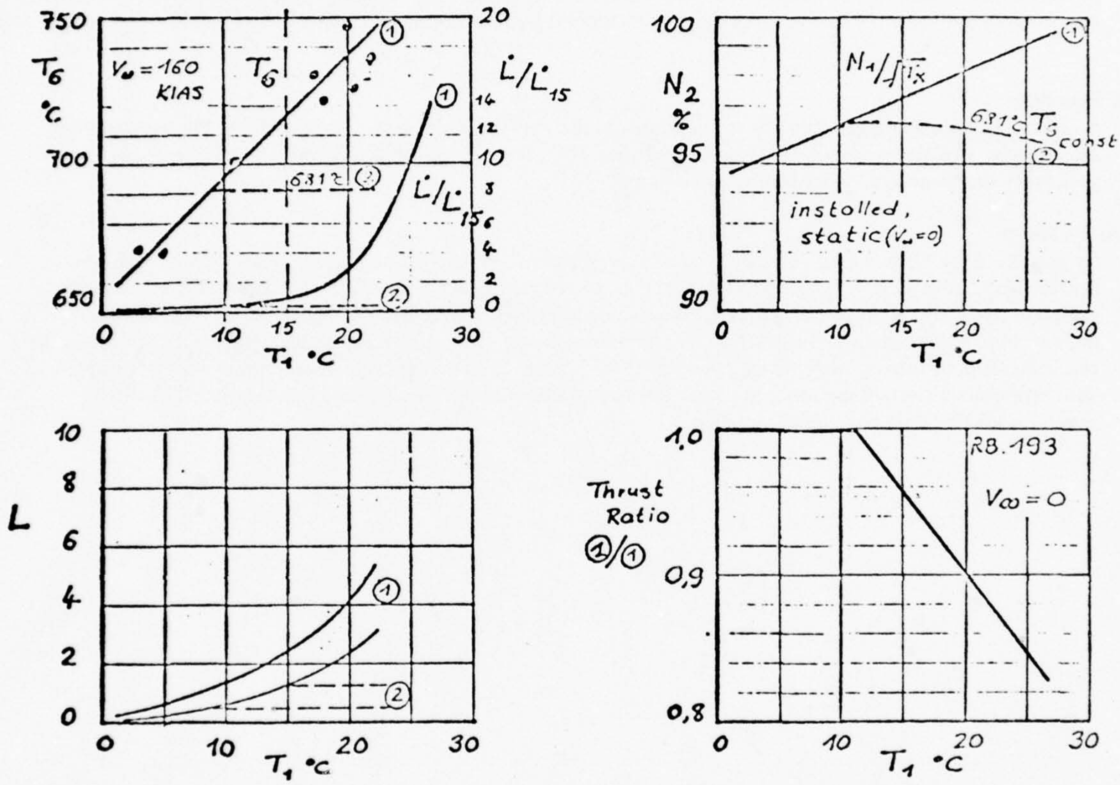


Fig.10 Typical in-service engine usage TGT pattern

DISCUSSION

R.Smyth

Turbine life usage can be monitored in a simple way by recording the turbine exit temperature and applying a life usage law to obtain turbine life used as a function of time. Although the absolute figure of turbine life used may be questionable it is a good measure of comparison for different types of engine usage to assess the severeness of life usage. Attached is a diagram showing the application of the above method for different rolling take-off procedures with a V/STOL fighter aircraft. On warm days maximum take-off power could be selected as either a corrected RPM or temperature limit.



T_1 = Inlet Temperature

T_6 = Exhaust Gas Temperature

L = Life Units per Flight

$\dot{L}$ = Rate of Life Unit Usage

$\dot{L}_{15}$ = $\dot{L}$ at T_6 corresponding to T_1 = 15°C

$N_1/\sqrt{T_X}$ = Engine Control Parameter

N_2 = RPM HP - Shaft

V_∞ = Flight Speed

Turbine life usage
during transition

Author's Reply

The problem of hot end life consumption is complicated by the fact that there is both creep and thermal fatigue occurring at the same time. Creep is a function of time spent at a temperature and thermal fatigue is a function of a number of accelerations and decelerations the engine sees and the maximum rpm of course that the engine sees. Although it is possible to predict the creep life of turbine blades, it has not been possible, I believe to predict with any accuracy the thermal fatigue for very high temperature engines with cooled turbine blades with complex cooling passages and complicated temperature distributions over the surface of the aerofoil. But I do agree with your original point that comparison between one engine and another on a creep base is sound. In the UK we are trying to move forward to a thermal life usage which embraces both creep and thermal fatigue and to indicate the usage of thermal life on some easily readable output display which in turn can be used to indicate the area and depth of inspection by borescope to verify the life recording reading.

H.I.H.Saravanamuttoo

I am referring to figure 10 of your presentation showing a gas temperature vs time. Which temperature do you use in your thermal life usage considerations? I believe that you have to use the combustor exit temperature and in case of complicated cooled turbine blades the blade temperature.

Author's Reply

The graph that was displayed was the estimated turbine gas temperature deduced from thermocouple measurements in the jet pipe and I agree with the speaker that this gives only a very rough indication of the metal temperatures of the turbine blades. However, in the UK developments of radiation pyrometers have proceeded to the point where we believe we have a tool which can be fitted to operational aircraft that will measure surface temperatures on turbine blades. In fact I have seen results which indicate that individual blades can be monitored by radiation pyrometer such that the circumferential temperature distribution is obtainable. I believe that in the future these instruments will form the basis of more realistic temperature monitoring of turbine blades and hot end components.

D.K.Hennecke

Since one never encounters uniform temperatures in the gas stream or on the blade surfaces the temperature sensors may pick up non-representative local values. What is your approach to tackle this old problem when you apply the measured values for life predictions?

Author's Reply

At present in the UK we have extensive research programmes on rigs and on engines which will indicate the turbine blade life and how it correlates with the measured parameters. Ideally those parameters will be from radiation pyrometers but useful estimates can also be obtained from jet pipe temperature measurements or inter turbine measurements. I agree that direct measurements of temperatures on blades are not themselves the sole indication of turbine life consumption but there is some chance that such measurements can be coupled with established design data, other known blade heat transfer characteristics, and related rig test results to provide a reliable "life-consumed" number.

DEVELOPPEMENT DU MOTEUR CFM56 ORIENTE VERS LA FIABILITE ET LA MAINTENABILITE

Monsieur Jean-Pierre MARECHAL
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SNECMA Direction Technique

77550 MOISSY-CRAMAYEL - FRANCE

L'exposé décrit les critères de fiabilité et de maintenance pris en compte dans le moteur CFM56 développé en commun par GENERAL ELECTRIC et la SNECMA dans le cadre d'un programme dirigé et commercialisé par la Société CFM INTERNATIONAL. L'objectif de ce programme est d'acquérir suffisamment d'expérience au cours du développement pour offrir aux utilisateurs dès la mise en service, un moteur adulte.Pour satisfaire à cet objectif, 7500 heures de fonctionnement sont prévues jusqu'à la certification du moteur dont une grande part sera consacrée à accumuler des cycles sévères représentatifs des conditions d'utilisation réelle d'un avion court-courrier. Parallèlement, pour satisfaire un coût d'entretien le plus faible possible par une rapidité et une facilité de remise en état, les impératifs de maintenance tels que modularité, accessibilité, interchangeabilité, ont été pris en compte dès le projet initial. L'accumulation d'observations systématiques au cours des essais permettra de mettre au point des méthodes de diagnostic et de mettre à la disposition des utilisateurs dès la mise en service,des guides de recherche de pannes.

GENERALITES

Le moteur CFM56 développé par GENERAL ELECTRIC et la SNECMA, dans le cadre d'un programme dirigé et commercialisé par la Société CFM INTERNATIONAL,est un moteur de 10 tonnes de poussée, de technologie avancée.

Double corps, double flux de taux de dilution élevé, le moteur CFM56 est conçu pour satisfaire les demandes des compagnies aériennes par une fiabilité accrue, une excellente maintenabilité, un rapport POUSSEE/POIDS élevé, une faible consommation spécifique, un niveau de bruit inférieur à celui défini par la nouvelle règlementation et un faible niveau de pollution.

Le moteur CFM56 est formé d'une part d'un ensemble haute pression possédant plus de 12 000 heures de fonctionnement à des températures d'entrée turbine supérieures à celles du moteur lui-même, cet ensemble étant utilisé sur le moteur F101, monté sur l'avion B1, et d'autre part d'un ensemble basse pression bénéficiant de l'expérience conjuguée des moteurs LARZAC, M45H, CF6-6 et CF6-50. Ce moteur de la nouvelle génération a fait l'objet d'une attention particulière pour faciliter et réduire les opérations de maintenance par un véritable découpage modulaire lié à des techniques de diagnostic développées parallèlement.

1. - DESCRIPTION SOMMAIRE DU MOTEUR

Le moteur CFM56 est une turbo-soufflante de taux de dilution de 6. Il possède un compresseur HP à géométrie variable et une turbine refroidie par air. Le rotor basse pression possède un étage de soufflante et trois étages de compresseur basse pression entraînés par une turbine à 4 étages. Le rotor haute pression possède 9 étages de pression entraînés par un seul étage de turbine. Les deux rotors sont supportés par 5 roulements sur 2 structures qui sont respectivement le carter intermédiaire supportant trois roulements et le carter d'échappement supportant le roulement arrière et indirectement le roulement inter-arbre, sur lequel s'appuie la turbine haute pression. La poussée est reprise au niveau du carter intermédiaire et le moment autour de l'axe du moteur peut être repris soit par l'une ou l'autre des structures.

L'étage de soufflante a des aubes à talon périphérique montées sur un disque anneau. Les trois étages mobiles de compresseur BP sont montés sur un tambour monobloc à queues d'aronde axiales, comme sur le M45H.

Le carter intermédiaire, principale structure du moteur, assure une rigidité suffisante pour conserver le contrôle des jeux rotor/stator en cas de charges limites telles que: efforts de reprise de poussée, charges aérodynamiques etc... Cette structure est conçue pour résister aux charges dues à la perte éventuelle d'une aube de soufflante.

La directrice de sortie de la soufflante est logée sur l'avant du carter intermédiaire.

Le compresseur HP se décompose en trois parties : les disques 1 et 2 groupés en forme de tambour, le rotor disque 3 à fixation séparée et un autre tambour supportant les étages 4 à 9. Les étages 1 - 2 et 3 utilisent des queues d'aronde axiales tandis que les aubes des étages 4 à 9 sont retenues dans des rainures circonférentielles comme sur le CF6. La directrice d'entrée et les redresseurs 1-2 et 3 à calage variable, sont actionnés par vérins hydrauliques. Le carter du compresseur est composé d'une partie avant en titane et d'une partie arrière en acier,chacune d'elles formée de 2 coquilles.

La turbine HP n'a qu'un seul étage dont les aubes fixes et mobiles sont refroidies par air. Dans les accouplements de la turbine HP et du compresseur HP sur l'arbre,sont interposés des disques comportant des léchettes de labyrinthes assurant un bon contrôle de l'étanchéité compresseur turbine.

La chambre de combustion très courte est annulaire à injection basse pression. Les émissions produites par cette chambre répondront à la règlementation en vigueur.

Le distributeur d'entrée de la turbine BP est refroidi et constitue un sous-ensemble par lui-même.

Le rotor de turbine BP à 4 étages est d'une conception à brides centrées par des boulons ajustés. Le rotor est entraîné par un arbre conique au niveau de la fixation des disques 2 et 3. Le carter de turbine d'une seule pièce est relié au carter d'échappement de forme polygonale.

Les enceintes avant et arrière pressurisées par de l'air inter-compresseur sont mises à l'air libre par un système séparateur qui chemine au centre de l'arbre BP et débouche au centre de la tuyère.

Le support des équipements est entraîné par le rotor haute pression, la prise de mouvement située au coeur du carter intermédiaire constituant un module entièrement indépendant. Le boîtier de transfert et le carter des équipements se situent à 6 heures, facilitant l'accessibilité sur avion. L'habillage est conçu pour satisfaire la décomposition modulaire du moteur et les accès aux éléments remplaçables en ligne (L.R.U.).

Parmi les moteurs à haut taux de dilution actuels, le CFM56 est celui qui comporte le moins de pièces, 34 % de moins que le CF6-50 par exemple.

2. - PROGRAMME DE DEVELOPPEMENT

2.1. - PLAN DU PROGRAMME ET ESSAIS ACCOMPLIS.

Le programme de développement du moteur CFM56, centré sur les essais de certification, a été conçu pour répondre aux objectifs de fiabilité, d'une part par des essais de survitesse, surchauffe, vibration et distorsion, prévus dans le cadre de la certification du moteur et d'autre part par une adaptation des cycles d'endurance aux réalités d'une mission type, voire même plus sévère, contribuant à valider des modifications bien avant la date de certification. Les techniques de maintenance telles que: interchangeabilité modulaire, validation de solutions de réparation, techniques de contrôle et suivi, sont également expérimentées sur les moteurs de développement.

Le premier moteur CFM56 a commencé ses essais en JUIN 1974. La pleine puissance a été atteinte seulement après 10 heures d'essais. En Octobre 1976, 2 000 heures de fonctionnement avaient été réalisées avec 6 moteurs dont deux sont en cours d'avionnage, l'un sur l'avion prototype A.M.S.T. YC15 de M.D. DOUGLAS, l'autre sur le banc volant CARAVELLE de la SNECMA. Contribuant aux essais dans le cadre de la certification, un grand nombre d'essais partiels a déjà été effectué ; nous citerons entre-autres les essais d'ingestion et de rétention qui ont permis de déterminer le comportement des composants soumis à des charges extrêmes et d'introduire dans certains cas des modifications bien avant la date de certification. A cet égard , la technologie de la partie frontale du moteur est directement issue de l'expérience acquise en essais réels. Une expérimentation globale, avec reproduction fidèle des paramètres moteur, a été effectuée dans un banc d'essais partiels. Plusieurs types de projectiles ont été employés :

- grêlons de ∅ 25 et 50 mm
- morceaux de glace 25x100x150 et 25x75x400 mm
- morceau de pneu
- oiseaux moyens 680 g (1,5 lbs)
- oiseaux lourds 1810 g (4 lbs)
- une aube de soufflante complète 2500 g (5,5 lbs).

Au total 52 tirs de projectiles ont été réalisés dont plus de 10 sur une roue de soufflante complète.

2.2. - OBJECTIFS

Les spécifications techniques du moteur en terme de performances, masse, fiabilité, maintenabilité, ont été établies à partir de l'expérience acquise sur les autres moteurs mais également en intégrant en permanence les dernières exigences des avionneurs et des compagnies aériennes. A cet égard, le moteur à déjà été présenté aux principaux avionneurs et compagnies aériennes mondiales, au cours de plusieurs réunions rassemblant la majorité d'entre eux.

Dans ce même cadre, l'objectif de durée de vie pour lequel le moteur a été conçu, correspond à un nombre considérable de cycles propre à chaque pièce principale , dans tous les cas la première dépose ne devant intervenir qu'au cours de la seconde moitié de la durée de vie totale de la pièce. La première condition pour les pièces tournantes, est remplie en s'assurant qu'un défaut de dimension donnée qui pourrait ne pas être détecté par les moyens de contrôle habituels, ne conduirait pas à une évolution à rupture avant la première dépose.

Le cycle commercial-type retenu correspond à un cycle d'une heure représentatif d'une utilisation intense sur un avion court-courrier donnant des conditions les plus sévères pour le vieillissement d'un moteur.

L'objectif est de reproduire les conditions d'amplitude des déformations de l'ensemble du moteur qui soient représentatives d'un cycle de vol type. Les parties chaudes et en particulier la turbine HP étant les plus critiques de ce point de vue, ce cycle a été obtenu sur les bases de réalisation suivantes :

- conditions de fonctionnement de la turbine haute pression
- températures obtenues sur le disque de turbine haute pression
- reproduction des amplitudes de déformation dues au gradient des températures mesurées.

La figure n° 1A schématise ce cycle commercial.

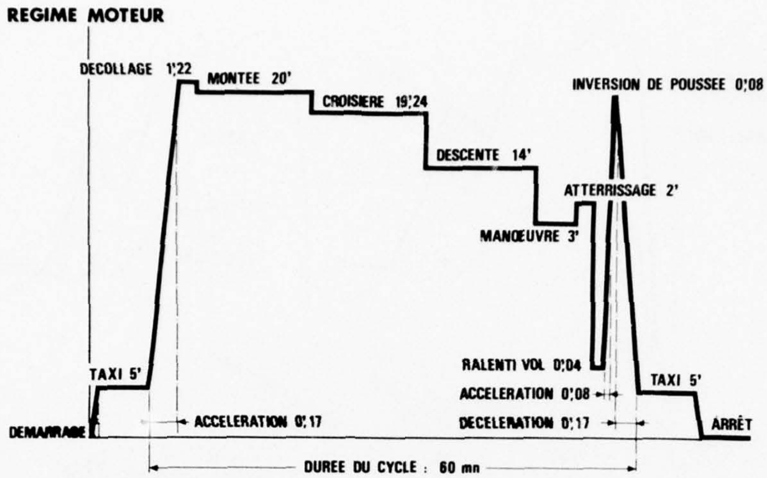

FIG. 1A

CFM 56 – MISSION COMMERCIALE TYPE

Le temps de montée constitue une part importante du cycle complet imposant des conditions de température qui affectent la durée de vie des composants en fatigue oligocyclique. La séquence en inversion de poussée a été également prise en considération pour tenir compte de la réduction et de la pointe de régime du moteur associées à cette manoeuvre.

2.3. - MOYENS MIS EN OEUVRE POUR PARVENIR A CES OBJECTIFS

Les moyens mis en place au cours du programme de développement permettent de s'assurer que les objectifs de durée de vie sont bien satisfaits mais également d'identifier les problèmes de fatigue afin de prévenir les pannes en service au cours des premières années d'utilisation. Cette analyse est faite à partir d'une combinaison de calculs, d'essais moteur et d'essais partiels.

Les critères pris en compte dans le dimensionnement sont :

- le fluage
- la fatigue oligocyclique
- la propagation de criques
- la survitesse
- les problèmes liés au flambage
- les fonctionnements anormaux détectés par l'analyse de pannes.

Pour tous ces critères, la méthode d'analyse tient compte du profil de vol de la mission commerciale-type, de la température ambiante et d'un niveau de dégradation du moteur en fonction du nombre de cycles.

Pour chaque critère, l'analyse se fait par étapes, appelées catégories qui sont :

CATEGORIE 5 : Analyse au niveau de projet.
CATEGORIE 4 : Analyse tenant compte des caractéristiques de matériaux obtenues par essais sur éprouvettes.
CATEGORIE 3 : Analyse de la catégorie 4 confirmée par les résultats aérothermodynamiques moteur.
CATEGORIE 2 : Analyse 3 complétée par les caractéristiques des matériaux obtenues sur éprouvettes attenantes.
CATEGORIE 1 : Analyse pour extension des performances du moteur.

Les moyens utilisés pour déterminer les concentrations de contrainte font appel à des techniques très sophistiquées telles que :

- analyse par éléments finis
- analyse de concentration de contrainte par photoélasticimétrie.

Cette méthode, très développée à la SNECMA pour l'étude des pièces fortement sollicitées (roues de turbine et de compresseur), permet de mesurer les contraintes et plus particulièrement les concentrations de contrainte difficilement accessibles au calcul. Elle utilise les propriétés élastiques particulières de l'araldite qui fige les contraintes consécutives à la mise en charge. L'examen en lumière polarisée permet de déterminer la direction et l'intensité des contraintes principales.

Les essais d'endurance sur moteur définis sur la base d'essais cycliques permettent de reproduire au mieux les conditions d'amplitude de déformation sur le fonctionnement de la turbine HP à partir du cycle de vol de la mission commerciale-type (figure 1B).

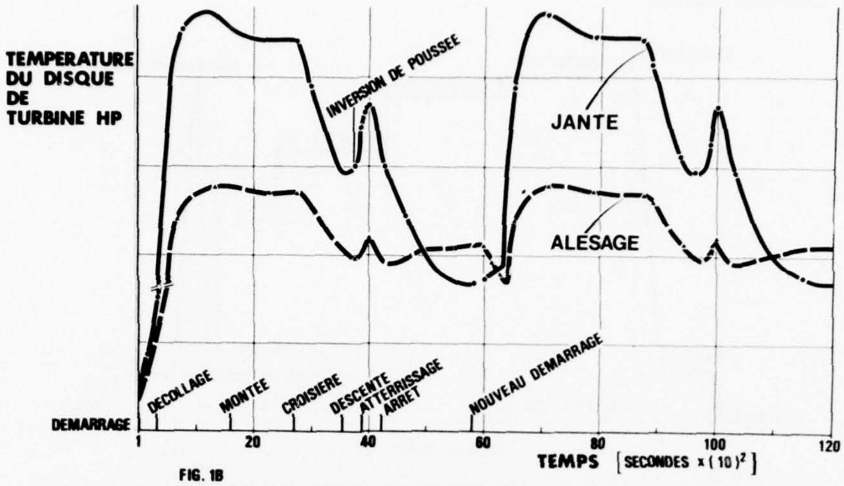

FIG. 1B

CFM 56._ REPARTITION DE TEMPERATURE SUR LE DISQUE HP DURANT LE CYCLE COMMERCIAL TYPE

Ces conditions tiennent compte :

- des gradients thermiques
- des charges aérodynamiques sur les arbres
- des charges sur les brides de fixation
- des températures réelles des matériaux
- des charges centrifuges.

Les figures 2A et 2B décrivent respectivement le cycle d'endurance et les températures obtenues sur le disque de turbine HP. Le temps d'arrêt du moteur entre deux cycles a été déterminé par les conditions de température entre la jante et l'alésage du disque HP, ce qui a pour effet de reproduire les amplitudes de déformation des contraintes maximales.

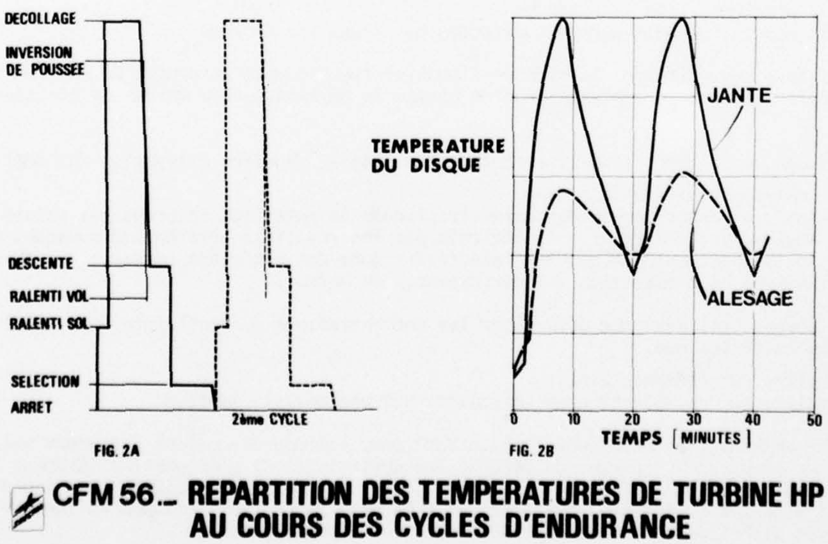

FIG. 2A

FIG. 2B

CFM 56._ REPARTITION DES TEMPERATURES DE TURBINE HP AU COURS DES CYCLES D'ENDURANCE

La comparaison avec les cycles d'endurance effectués sur le moteur CF6 montre que le cycle choisi pour le CFM56 (Figure 3) conduit à :

- des températures de régime de décollage durant une période plus longue
- des conditions plus sévères, imposées par le maintien d'un niveau de poussée plus élevé après l'inversion de poussée
- des transitoires plus sévères
- un temps de refroidissement entre deux cycles plus long ayant pour effet d'augmenter les gradients thermiques.

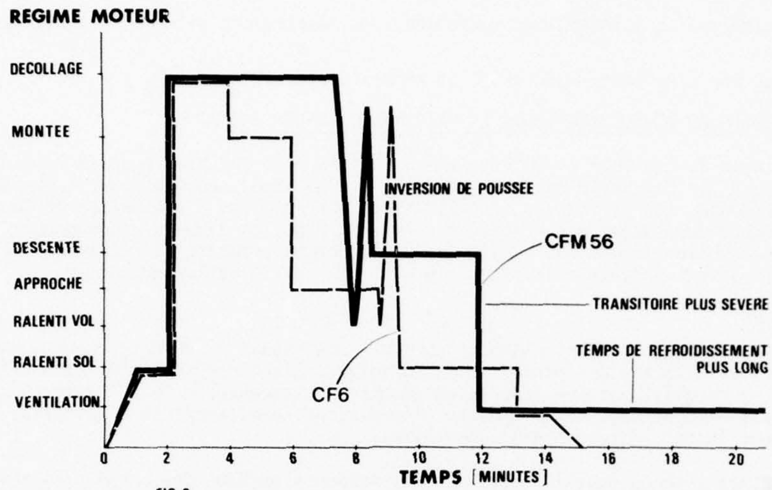

FIG. 3

CFM56._COMPARAISON AVEC LE CYCLE D'ENDURANCE DU CF6

De plus au cours de ces essais cycliques, il est prévu de faire fonctionner le moteur avec des balourds supérieurs à ceux normalement tolérés, à des températures de composants plus élevées que celles que l'on rencontrera en utilisation normale et avec les équipements fonctionnant au-delà de leur charge nominale.

A la certification 11 500 cycles d'endurance auront été réalisés et devront permettre d'éviter les problèmes à la mise en service du CFM56 ou d'en limiter les conséquences.

3. - OBJECTIFS DE MAINTENANCE

Parallèlement aux efforts mis en oeuvre pour améliorer la fiabilité du moteur, les équipes de maintenance ont participé à l'élaboration et au développement des concepts de maintenance tant en ce qui concerne les critères d'accessibilité et d'interchangeabilité, que les méthodes de suivi de la santé du moteur.

Cette approche est menée à bien par les méthodes suivantes :

- Revue détaillée des dessins du moteur permettant d'analyser par un diagramme logique de maintenance, le démontage du moteur.

  Cette analyse couvre tous les niveaux de maintenance, et est suivie par une analyse technique de la conception du moteur sous l'angle maintenabilité, permettant de déterminer le degré de crédibilité en fonction des objectifs fixés. Cette étude a commencé très tôt dans le programme de développement afin d'incorporer les modifications de conception au cours des différentes versions du moteur, de prévoir les études d'outillages et de faciliter les opérations de réparation.

- Analyse logique des pannes potentielles permettant une optimisation du programme de maintenance (MSG). Cette étude, développée par les compagnies aériennes sur le BOEING 747, est devenue une procédure obligatoire pour la constitution et la présentation du programme de Maintenance. Elle a pour but de vérifier en détail et dans l'ordre, les points suivants :

  - identifier les systèmes
  - identifier les pannes et leurs effets
  - déterminer les actions à mener en ce qui concerne la sécurité ou les fonctions cachées
  - déterminer parmi toutes les possibilités, celles qui sont les plus économiques.

- Vérification tout au long du développement sur les moteurs en essais.

  a) de l'interchangeabilité modulaire en essayant d'utiliser au plus tôt les outillages de série de façon à les valider

b) des solutions de réparation y compris le reconditionnement des revêtements de protection

c) des durées des différentes interventions réalisables sur un moteur avionné.

Comme déjà souligné précedemment la présentation du moteur aux compagnies aériennes a permis d'établir un échange fructueux entre utilisateurs et concepteurs dans les domaines de performances, des systèmes, de la technologie ou de la maintenance. Les compagnies aériennes ont été tenues informées des modifications apportées sur le moteur tout au long de l'avancement du programme. GENERAL ELECTRIC et la SNECMA ont incorporé environ une centaine de modifications dont certaines résultent des différentes réunions avec les compagnies aériennes américaines et européennes.

Les deux concepts de base qui ont été adoptés sont :

- de substituer à la maintenance programmée, une maintenance selon l'état utilisant des techniques de suivi.
- d'assurer une très bonne modularité et réparabilité.

3.1. - TECHNIQUES DEVELOPPEES POUR LA MAINTENANCE SELON L'ETAT

Les techniques de contrôle et suivi ont été conçues pour satisfaire aussi bien une utilisation manuelle qu'automatique par des systèmes embarqués tels que A.I.D.S. Les équipements et dispositifs techniques ainsi que toutes les procédures d'utilisation sont en cours d'évaluation et font partie intégrante du développement, une partie essentielle est la recherche des signatures de pannes. Toutes les signatures de pannes établies durant les essais de développement permettront d'élaborer les guides de recherche de pannes qui seront fournis aux utilisateurs dès la certification du moteur.

SUIVI DES PERFORMANCES.

Une méthode de diagnostic est élaborée au cours des essais de développement à partir des paramètres sélectionnés pour la vérification des performances du moteur. L'analyse réalisée selon cette méthode pourra être faite soit en vol, en utilisant les indications données à la cabine ou par l'intermédiaire de systèmes embarqués, soit au sol en utilisant des sondes complémentaires, améliorant le diagnostic. Le moteur comprend tous les bossages pour installer les sondes nécessaires.

Les paramètres retenus pour le suivi des performances en plus des paramètres obligatoires sont :

- pressions amont et aval soufflante
- pression et température d'entrée compresseur HP
- pression et température de sortie compresseur HP
- pression et température de sortie turbine BP

La méthode de diagnostic, basée sur l'évolution de coefficients d'influence, mise au point au cours du développement, permettra d'isoler le module éventuellement en cours de détérioration, pouvant affecter les performances. Les guides de recherche de pannes seront élaborés en tenant compte des constatations sur les pièces moteur après essais. La possibilité d'isoler les modules affectant les performances permettra aux utilisateurs de connaître à tout moment l'état de l'ensemble du parc pour éventuellement programmer à l'avance les opérations de dépose moteurs ou entreprendre les actions correctives sur l'aile, ou encore de donner les précisions sur les réparations nécessaires en limitant l'indisponibilité de l'avion.

SUIVI VIBRATOIRE DU MOTEUR.

Le suivi vibratoire des moteurs sur les avions actuels n'a pas toujours répondu aux besoins des utilisateurs, en raison particulièrement de la fiabilité des accéléromètres, et des problèmes liés à la transmission des signaux. Sur le CFM56, le développement de la chaîne de vibration fait partie intégrante du programme moteur et un soin particulier a été apporté sur l'amélioration de la fiabilité de tous les éléments composant la chaîne.

L'emplacement et les caractéristiques des capteurs ont été choisis à partir de l'expérience acquise avec les mesures vibratoires faites sur le moteur pour les besoins du développement.

Le moteur comportera les aménagements permettant d'incorporer les capteurs piezo-électriques de grande sensibilité sur les principaux paliers du moteur et le support des équipements.

En dehors du suivi de la tranquilité de marche du moteur, la chaîne de mesure des vibrations permettra de détecter :

- des balourds de toutes les parties tournantes, et par conséquent les dégradations de performances résultant de l'augmentation des jeux entre rotors et stators
- des avaries de paliers
- des dégradations de pignons, roulements de l'ensemble de la chaîne cinématique
- des défectuosités d'équipements.

De plus, elle permettra d'effectuer les opérations d'équilibrage dynamique de la soufflante.

Les guides de recherche de pannes donneront les signatures de pannes observées au cours du développement ainsi que les consignes aux utilisateurs sur les valeurs des charges dynamiques au-delà desquelles le potentiel des pièces moteur est affecté.

SURVEILLANCE DU CIRCUIT D'HUILE

Le circuit d'huile du CFM56 comporte une récupération séparée pour chaque enceinte sur laquelle est incorporé un piège à particules comprenant, un bouchon magnétique et un filtre dans un ensemble self-obturateur.

Cette fonction est assurée par le module de lubrification qui groupe la totalité des pompes et facilite les opérations de maintenance. Cette méthode permet de surveiller l'état des pièces baignées par l'huile, et le guide de recherche de pannes donnera toutes les indications permettant d'isoler la pièce éventuellement détériorée.

Associée à cette méthode, l'analyse spectrométrique de l'huile permet de donner des renseignements complémentaires tels que la vitesse de pollution, renseignant sur les usures lentes des pièces soumises au frottement. L'orifice de prélèvement prévu pour l'analyse spectrométrique se situe sur le réservoir d'huile et est conçu pour être utilisé sans outillage particulier.

EXAMEN ENDOSCOPIQUE

Le moteur CFM56 a fait l'objet d'une attention particulière sur le nombre des orifices endoscopiques. 23 orifices sur l'ensemble du moteur sont prévus, qui se décomposent comme suit :

- 1 sur le compresseur BP
- 9 sur le compresseur HP
- 6 sur la chambre de combustion et la turbine haute pression
- 7 sur la turbine basse pression dont 4 sur le premier distributeur.

Des endoscopes de diamètre égal à 10 mm pourront être utilisés sur la plupart des orifices, permettant un examen suffisamment précis dans le cadre des opérations de maintenance en ligne.

L'excellente accessibilité des orifices permettra l'utilisation de systèmes télé-endoscopiques, améliorant les conditions de détection de défaut et réduisant les temps d'inspection.

Une procédure complète,mise au point au cours du développement, sera fournie aux utilisateurs avec les guides de recherche de pannes correspondants.

EXAMEN GAMMAGRAPHIQUE

L'examen gammagraphique du CFM56 permet de vérifier les parties internes du moteur sans désassemblage. Cette opération est facilitée par la possibilité d'introduire soit par l'avant, soit par l'arrière le tube porte-source à l'intérieur de l'arbre, après dépose d'un simple obturateur.

Cet examen complète les résultats des autres méthodes d'analyse avant une intervention sur le moteur.

Des essais sont répartis au cours du programme et permettront de démontrer les possibilités de l'examen gammagraphique sur l'ensemble du moteur et d'établir les procédures d'utilisation et les images des avaries type.

3.2. - CONCEPTION MODULAIRE DU MOTEUR.

L'efficacité de la maintenance a été réellement améliorée par une conception modulaire des moteurs associée à des méthodes de diagnostic plus rigoureuses.

La conception du moteur CFM56 correspond à une seconde génération de ce point de vue. Les modules constituent des ensembles indépendants à un niveau suffisamment petit pour satisfaire les différentes options possibles des utilisateurs. L'interchangeabilité modulaire mécanique et fonctionnelle a été soignée et de plus, est facilitée par l'utilisation de la technique de l'équilibrage modulaire. Le moteur CFM56 possède quatre ensembles majeurs qui sont :

- l'ensemble de soufflante
- le corps haute pression
- la turbine basse pression
- le carter des équipements.

Chacun de ces ensembles est constitué d'un nombre de modules qui peuvent être associés entre eux suivant les désirs de l'utilisateur.

L'ensemble de soufflante est composé de 4 modules qui sont respectivement, le rotor de soufflante compresseur BP, le support de palier 1 - 2, la structure de soufflante, la prise de mouvement.

Le corps haute pression est composé de 7 modules. Le rotor de turbine HP, la chambre de combustion et le distributeur HP peuvent être remplacés sur l'aile. Les carters des compresseurs en deux parties procurent une accessibilité exceptionnelle.

La turbine BP se compose de 4 modules ; le distributeur constituant à lui seul un module, ce qui permet le remplacement de secteurs du distributeur sans désassembler la turbine BP.

Les principales caractéristiques de conception facilitant les opérations de maintenance sont :

- les aubes de soufflante remplaçables par paires
- les aubes du redresseur de soufflante remplaçables individuellement
- des aménagements pour l'équilibrage, très aisément accessibles sur toutes les pièces tournantes
- des surfaces de référence prévues sur toutes les pièces pouvant nécessiter une réparation
- des surépaisseurs pour la réparation sur toutes les brides.

4. - CONCLUSION

Après plus de trois années de coopération entre GE et la SNECMA, le programme de développement du moteur CFM56 est respecté : 2045 heures de fonctionnement ont été réalisées à la date du 30/10/76 pour 1945 heures prévues. Les modifications apportées sur les moteurs 005 et 006 à partir de l'analyse des essais partiels ou complets des moteurs, contribuent à l'approche de la définition du moteur de série.

Les essais d'endurance combinés à l'analyse de fiabilité des composants du moteur CFM56, associés aux méthodes de contrôle de l'état de santé du moteur,concourront à diminuer la fréquence des interventions de maintenance dès sa mise en service.

CFM56 TURBOFAN MAINTAINABILITY AND RELIABILITY - ORIENTED DEVELOPMENT

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This paper describes reliability and maintainability criteria introduced in the CFM 56 turbofan engine, jointly developed by General Electric and SNECMA within the framework of the program managed and marketed by CFM International. The objectives of this program is to accumulate enough experience through development to provide operators with a mature engine as soon as entry into service. To meet this objective, 7500 operating hours are planned before engine certification, a large part of this running time being assigned to accumulation of severe cycles providing a realistic simulation of actual service. Simultaneously, to achieve minimized maintenance costs through high repairability, accessibility, modularity and interchangeability features have been considered at the early design stage. Through a methodical accumulation of data observed, diagnosis procedures will be developed, and trouble shooting guides will be made available to operators as soon as the engine enters commercial service.

GENERAL

The CFM56 engine developed by General Electric and SNECMA, within the framework of a program managed and marketed by CFM International is an advanced turbofan in the 10-ton class. This high bypass twin shaft turbofan has been designed to meet the airline requirements through an improved reliability, an out-standing maintainability, a high thrust-to-weight ratio, a low specific fuel consumption, a noise level below the new regulation limits and a low pollution level.

The CFM56 combines first a core logging more than 12000 hours of operation at turbine inlet temperatures even higher than the CFM56 level - this is the core used in the F 101 engine, powering the B1 aircraft - and, second, a low pressure system using the experience gained from LARZAC, M45H, CF6-6 and CF6-50 engines.

This new generation engine has been the subject of a very careful attention to facilitate and reduce maintenance operations, through a real modular breakdown associated to simultaneously developed failure diagnosis techniques.

1 - SUMMARIZED DESCRIPTION

The CFM56 is a 6 to 1 bypass ratio turbofan. This engine features a variable stator HP compressor, and an air-cooled turbine. The low pressure rotor system includes one fan stage and a three-stage LP compressor driven by a four-stage turbine. The high pressure rotor system includes a nine-stage compressor driven by a single-stage turbine. The two rotor systems rotate on five bearings supported by two main frames, namely the fan frame, incorporating three bearings, and the turbine frame, incorporating the rear bearing and the inter-shaft bearing supporting the HP turbine. Thrust loads are taken at the fan frame, and moment loads about the engine centerline can be taken by either main frame.

The fan stage incorporates tip-shrouded blades fitted to an annular disc. The three LP compressor blade stages are installed in axial dovetail slots machined in a drum, similar to the M45H arrangement.

The fan frame, which is the major structural component of the engine, has the adequate inherent stiffness required to control rotor/stator clearances in limit load conditions - thrust loads, aerodynamics loads, etc... This structure has been designed to withstand stresses resulting from a fan blade loss. The fan outlet guide vane assembly is fitted at the fan frame front end.

The HP compressor is made in three sections : rotor stages 1 and 2 as an integral drum, rotor stage 3 as an individual disk, and a rear drum for stages 4 to 9. Dovetail blade attachment is used on stages 1-2 and 3, while a peripheral slot design similar to the CF6 configuration is used on stages 4 to 9. Inlet guide vanes and stator stages 1-2 and 3 are variable, and controlled by hydraulic jacks. The compressor casing front section is made of titanium, and the rear section of CR.Ni Alloy each section being designed as two half-shells.

The single stage HP turbine blades and vanes are air-cooled. Seal-support disks are mounted from HP compressor and turbine to shaft couplings to provide a satisfactory control of leaks between the compressor and the turbine.

The combustor is a very short low pressure injection, annular design. The emission level from this combustor meets the applicable regulations.

The LP turbine nozzle guide vane assembly is air-cooled and constitutes a complete individual unit.

The four-stage LP turbine rotor is a bolted-flange design, flange location being achieved by fitted bolts. The turbine rotor is driven by a conical shaft from the joint flange between stage 2 and 3 disks. The one-piece turbine casing is attached to the polygonal turbine frame.

Forward and aft sumps are pressurized with air bleed from the HM compressor inlet and through a center-vent tube.

The accessory gearbox is driven from the core engine rotor. Inlet gearbox, located right in the fan frame center, forms a fully separate module. The transfer gearbox and accessory gearbox are located at the 6 o'clock position to facilitate installed accessibility. External dressing items have been laid out to suit the engine modular breakdown and accessibility to line replacement units.

From current high bypass turbofans, the CFM56 is the engine incorporating the smallest number of components : 34 % less than the CF6-50, for instance.

2 - DEVELOPMENT PROGRAM

2.1 - PROGRAM PLAN AND TEST STATUS

Based on certification testing, the CFM56 engine development program has been prepared to meet reliability objectives by overspeed, overtemperature, vibration and distorsion tests scheduled in the certification program and by adapting endurance cycles, to the actual conditions of an even more severe typical mission contributing to clearing design changes long before certification.

Maintenance techniques such as modular interchangeability, clearance of repair schemes, monitoring techniques are also tested on development engines.

Initial tests of the first CFM 56 engine took place in June 1974. Full power was reached after 10 hours of testing only. In October 1976, 2000 hours of operation had been logged with 6 engines. Two of these are currently being installed on two aircraft : the M.D. Douglas YC 15 AMST prototype and SNECMA's Caravelle Flying test bed. As part of certification work, a large number of rig testing has been already carried out, including containment and foreign body ingestion tests which provided information on integrity of components submitted to extreme loads, and, in some instances, have resulted in the application of design changes well ahead of the certification date.

In this connection, the engine front end mechanical design is directly issued from actual tests. Global testing, with true reproduction of engines parameters, has been carried out in a component test rig. Several projectile kinds have been used :

- hailstones 1 and 2 in.
- ice slabs (1 x 4 x 6 in.) and (1 x 3 x 16 in.)
- medium birds (1,5 lb)
- large birds (4 lb)
- tire thread

52 shots have been made, 10 of which on a complete fan wheel. In addition, a containment test has been carried out for the first time on an engine with deliberate failure of one fan blade at the root.

2.2 - OBJECTIVES

The engine technical requirement in terms of performance, weight, reliability, maintainability have been established on the basis of the background experience gained on other engines, and also by introducing the latest aircraft manufacturer and airline requirements. In this connection, the engine has already been presented to major aircraft manufacturers and airlines during several meetings.

In the same context, the CMF56 design life for the major engine components corresponds to a considerable number of cycles : in all cases, the first removal must not be required before the second half of the overall component life. For rotating components, the first requirement is met by making sure that a pre-determined size defect which could remain undetected by usual inspection methods will not propagate to failure before the next removal.

The commercial cycle selected corresponds to a one-hour cycle representative of typical short range operation, and leads to the most adverse conditions as far as engine ageing is concerned.

The objective is to reproduce strain range conditions of the whole engine, representative of a typical flight cycle. Hot section components and, in particular, the HP turbine being the most critical components from this point of view, the cycle has been based on the following conditions being met :

- HP turbine operating conditions
- HP turbine disk temperature
- reproduction of strain range corresponding to measured temperature gradient.

This flight cycle is shown by figure 1A.

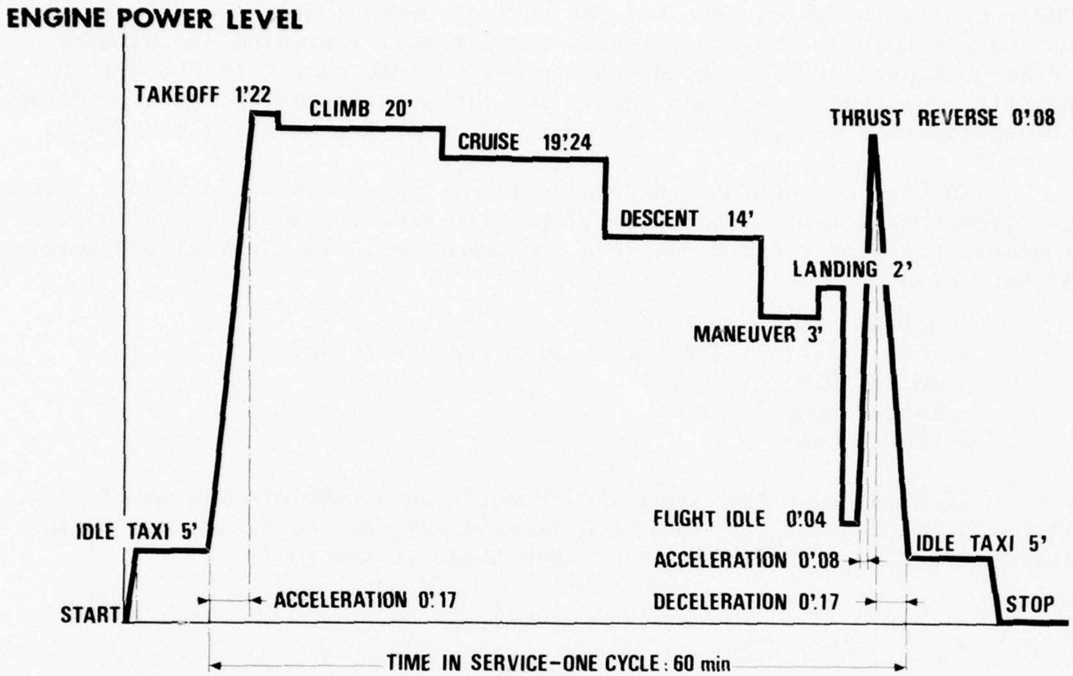

FIG. 1A

CFM 56 — TYPICAL COMMERCIAL CYCLE

Climb constitutes a large portion of the complete cycle, corresponding to temperature conditions affecting the component low cycle fatigue life. Reserve thrust sequence has also been considered to introduce the engine deceleration and re-acceleration related to thrust reverser selection.

2.3 - MEANS USED TO MEET THESE OBJECTIVES

Means used during the development program will provide confirmation that life objectives are met, and identification of fatigue problems in order to prevent in-service failures during initial years of service. This analysis is based on the combination of calculation and test data.

The following criteria are included in the design :

- creep
- low cycle fatigue
- crack growth
- overspeed
- buckling
- functional troubles detected by failure analysis.

For all these criteria, the analysis method considers the typical commercial mission profile, ambient temperature, and engine degradation with the number of cycles.

For each criteria, this analysis includes several steps corresponding to the following categories :

Category 5 : Analysis at the project design stage

Category 4 : Analysis introducing material property data obtained on test pieces

Category 3 : Analysis to category 4 confirmed by aero-thermodynamic engine data

Category 2 : Analysis to category 3, with additional material property data obtained on attached test pieces

Category 1 : Engine performance growth analysis.

Highly sophisticated techniques - e.g. finite element analysis and photoelasticimetry are applied to investigate stress concentrations. Photoelasticimetry, a method widely developed by SNECMA to investigate highly stressed parts, is used to measure stresses and, in particular, stress concentrations which cannot easily be predicted by calculation. This method is based on special properties of araldite, whereby stress patterns can be "frozen" after loading. Analysis of such patterns under polarized light shows up direction and magnitude of main stresses.

Engine endurance tests defined on the basis of cyclic tests will provide the best possible simulation of strain range effects on HP turbine operation in flight conditions corresponding to the typical commercial mission (see Fig. 1B).

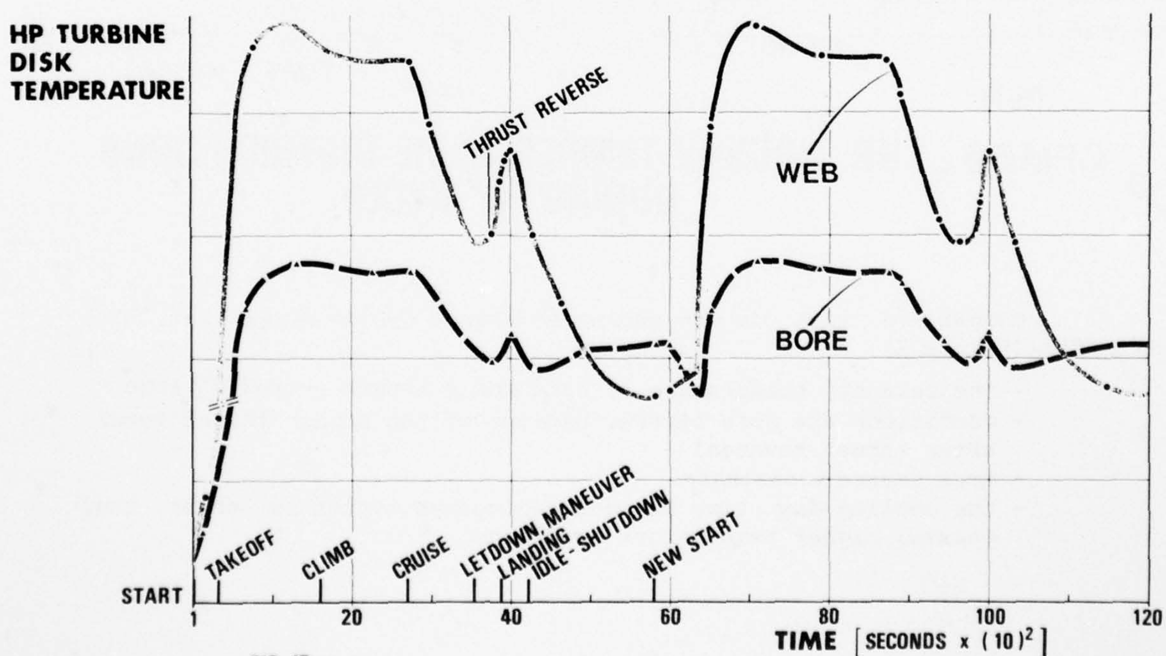

FIG. 1B

CFM 56 — LIFE VERIFICATION

These effects include :

- temperature gradients
- aerodynamic load on shafts
- loads on attaching flanges
- actual material temperatures
- centrifugal loads.

The endurance cycle and temperatures measured on the HP turbine disk are shown by figures 2A and 2B respectively. The engine shutdown time between two successive cycles has been established on the basis of temperature conditions between the HP disk web and bore, which consequently will reproduce the strain range of maximum stresses.

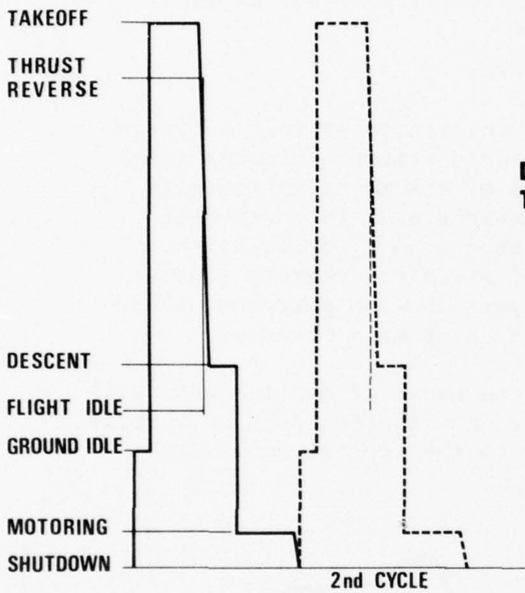

FIG. 2A

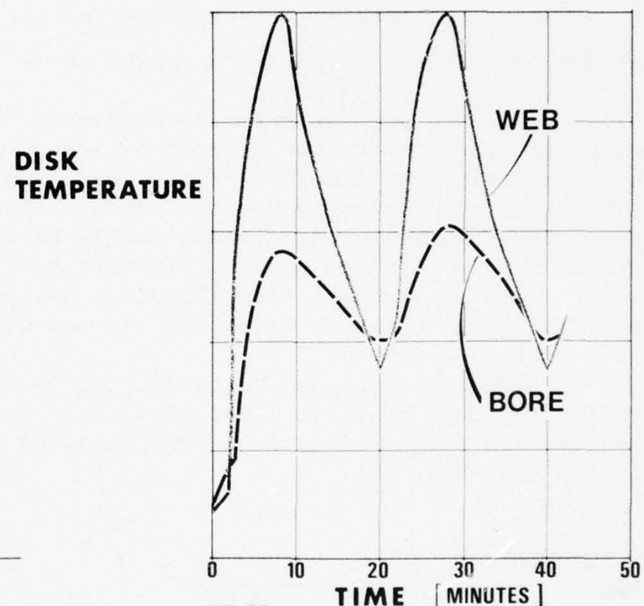

FIG. 2B

CFM56 – HP TURBINE TEMPERATURE DISTRIBUTIONS DURING "C" CYCLE

Comparison with the CF6 engine endurance cycles shows that, with the CFM56 (figure 3)

- the take off temperature is used for a longer period of time
- conditions are more severe, because of the higher thrust level after thrust reversal
- more severe transients
- the cooling down time between successive cycles is longer, thus causing higher temperature gradients.

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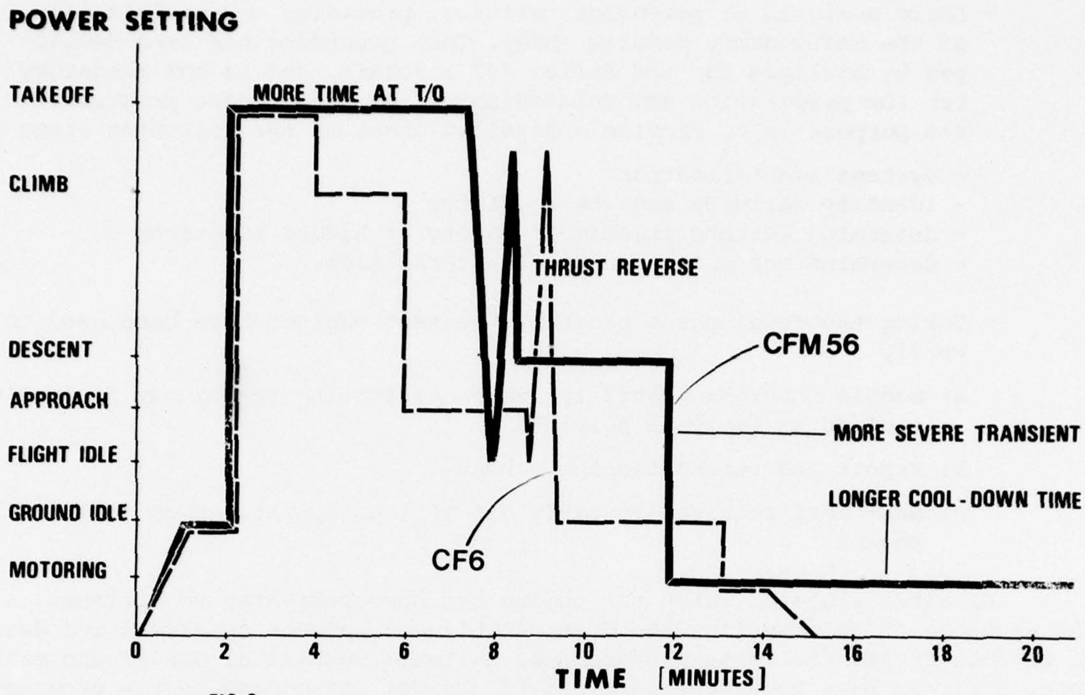

FIG. 3

CFM56 – SIMULATED SERVICE CYCLE ("C" CYCLE)

In addition, these cyclic tests will include engine running with unbalance levels higher than those normally acceptable and with component temperatures above those of normal service.

At certification, 11500 endurance cycles which will have been made, should provide solution to problems expected in service.

3 - MAINTAINABILITY OBJECTIVES

Simultaneously with work made to improve engine reliability, maintenance teams took part to design and development of maintenance concepts both in terms of interchangeability and accessibility criteria, and of engine health monitoring methods.

This exercice is being successfully developed through the application of the following methods :

- Detail review of engineering drawings to analyse engine disassembly sequence through a maintenance logic diagram.

  All maintenance levels are covered by this analysis, which is followed by a maintainability oriented engine design analysis to determine how realistic the objectives are. This exercise has been started quite early in the program, in order to introduce design changes progressively in the various engine versions, to prepare tooling studies and to facilitate repair operations.

- Logic analysis of potential failures, providing an optimization of the maintenance program (MSG). This procedure has been developed by Airlines for the Boeing 747 aircraft, and is now mandatory for the preparation and submission of the maintenance program. Its purpose is to provide a detailed check of the following items :
  - systems identification
  - identify failures and their effects
  - determine actions related to safety or hidden functions
  - determine the most economical alternatives.

- During the development program, the test engines have been used to verify :

  a) module interchangeability, while attempting to try out production tooling as early as possible

  b) repair and reconditioning schemes

  c) man-hours required to carry out various operations on an installed engine.

As already stated, when the engine has been presented to airlines, a useful exchange of information has been established between operators and designers in such varied fields as performance, systems, mechanical design and maintenance. Airlines have been kept informed of changes introduced as the program was proceeding. General Electric and SNECMA have introduced about a hundred changes some of which were the result of various meetings held with U.S. and European Airlines.

The two basic concepts adopted are :

- scheduled maintenance replaced by on-condition maintenance supported by monitoring techniques

- to provide a high degree of modularity and repairability.

3.1 - TECHNIQUES DEVELOPED FOR ON-CONDITION MAINTENANCE INPLEMENTATION

Monitoring techniques have been developed to meet both the requirements of manual and automatic on-board systems - A.I.D.S. - Monitoring equipment and operating procedures are being evaluated as an integral part of the engine development program. Identification of failure signatures is an essential part of this exercice. All failure signatures identified during development test will be used to prepare trouble-shooting guides available for operator's use as soon as engine certification is obtained.

PERFORMANCE MONITORING

A diagnosis method is worked out during development tests from parameters selected for performance monitoring. Engine performance analysis based on this method can be made either in flight from flight deck data or by means of on-board systems, or on the ground by means of additional probes in order to achieve a more detailed diagnosis. The engine incorporates all bosses required to fit the appropriate probes.

The following parameters have been selected for performance monitoring in addition to data mandatorily used :

- fan inlet and outlet pressure
- temperature and pressure at core compressor inlet
- temperature and pressure at core compressor delivery
- temperature and pressure at LP turbine delivery.

The diagnosis method, based on the variation of coefficients of influence, established during the development program will enable any faulty module to be identified. Trouble shooting guides will be prepared on the basis of engine component condition subsequent to testing. Through identification of modules, operators will be able to know at any time the condition of all engines, and to predict engine removals or to make "on wing" corrective actions, or to provide information on repairs required with a minimized effect on aircraft availability.

VIBRATION MONITORING

On current aircraft, engine vibration monitoring has not always met operators requirements, this being mainly due to sensors reliability and problems associated with signal transmission. On the CFM56, the vibration instrumentation system development is integrated in the engine program, and improvement of system components reliability has been very carefully considered.

Sensors lay-out and performance have been selected from experience acquired with vibration data obtained from engine mounted sensors during development tests.

The engine will incorporate provisions for high sensitivity vibration cristal sensors on main engine bearings and accessory gearbox.

In addition to engine vibration, the following conditions will be monitored by the vibration instrumentation system

- rotor unbalance and therefore performance deterioration due to increased clearances
- bearing failures
- wear of gears and bearings of the accessory drive system
- accessory malfunctions.

Also, the vibration monitoring instrumentation will facilitate trim balance operations.

Trouble shooting guides, together with failure signatures recorded during the development program, will provide operators with information about dynamic load levels beyond which component life is affected.

LUBE SYSTEM MONITORING

The CFM56 lube system incorporates a separate scavenge circuit for each sump. Such circuit includes a "particle trap" made of a magnetic plug and a filter housed in a self-sealing assembly.

The scavenging function is achieved by the lubrification module grouping all pumps in a single unit for maintenance simplification. With this system condition of lube-wetted components can be monitored and trouble shooting guides will provide all information required to identify any damaged component.

Spectrometric oil analysis, associated to the above method will generate additional data such as contamination rate, providing an information on component wear. The lube tank is provided with a sampling adaptor requiring no special tooling.

BORESCOPE INSPECTION

The number of borescope inspection bosses on the CFM56 engine has been the subject of a careful consideration. The engine will incorporate 23 borescope bosses :

- 1 on the booster
- 9 on the core compressor
- 6 on combustor and HP turbine
- 7 on LP turbine, including 4 on the first stator stage.

Most of these bosses are suitable for .4 in. diameter borescope, providing for an adequate level of checking accuracy for field maintenance operation.

Easy accessibility to these bosses will enable the use of borescopes associated with video systems, to improve defect detection and reduce inspection time.

A comprehensive procedure tried out during the development program will be made available to operators, together with a trouble shooting guide.

GAMMA-RAY INSPECTION

Gamma-Ray inspection will be used to check the CFM56 internal components without any removal. Design features have been introduced to allow the introduction of the source from the front or from the rear after removing a cover.

This type of check comes as an addition to the other analysis methods to provide data before making any action on the engine.

Specific tests are included in the program to demonstrate how gamma-ray inspection can be used throughout the engine, and to derive the relevant operating procedures and produce radiographs of typical defects.

3.2 MODULAR INSPECTION

Efficiency of maintenance has been greatly improved through a modular design, associated to stricter diagnosis methods.

The CFM56 mechanical design corresponds to a second generation from this point of view. Modules constitute separate assemblies sufficiently small to meet operators options. In addition, functional and mechanical interchangeability has been very carefully considered and, moreover, is facilitated by the modular pre-balancing technique. The CMF56 engine incorporates the following four major assemblies.

- fan assembly
- core engine
- low pressure turbine
- accessory gearbox.

Each major assembly includes several modules which can be associated as required by the operator.

The fan system comprises 4 modules : fan/booster rotor, bearing n° 1 and n° 2 support, fan frame and inlet gearbox.

The core system comprises 7 modules. The HP turbine rotor, the combustor and the turbine nozzle can be replaced "on the wing". Compressor casings, made in two halves, provide a high degree of accessibility.

The LP turbine comprises 4 modules : the turbine nozzle is itself one module, which allow the replacement of vane sectors without having to disassemble the LP turbine.

The main design features improving maintenance operations are :

- fan blades replaceable in pairs
- fan stator vanes replaceable individually
- high accessibility balancing provisions on all rotating components
- datum surfaces on all items which may required to be repaired
- repair stock allowances on all flanges.

4 - CONCLUSION

After more than three years of cooperation between General Electric and SNECMA, the CFM 56 development program is proceeding as scheduled. On october 30th 1976, 2045 hours of testing have been made, against a scheduled figure of 1945. The design changes introduced in engines 005 and 006 from the analysis of rig or engine tests are contributing to reaching the final definition of production engines.

Endurance tests, combined with component reliability analysis and engine health monitoring will reduce maintenance operation frequency as early as entry into service.

Aircraft Engine Design & Development Through Lessons Learned

by

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During the past twenty five years the industry has progressed from simple, single spool, low pressure ratio turbojet engines to complex, multi-spool, high pressure ratio, high temperature turbofans used to power huge wide bodied aircraft like the C-5A, DC-10, and 747. Engine life and reliability have improved steadily over that in the era of reciprocating engine aircraft. These have produced a major favorable impact on airworthiness and maintenance cost for commercial carriers. From this the question arises, "What does it take to design and develop a modern aircraft engine?" Engineers who worked on the early engine programs were just as intelligent as we are today, but they did not have the materials, modern analytical and experimental techniques, and the "lessons learned" from experience to guide them. The aircraft gas turbine engine has an almost indeterminate number of potential failure modes. Learning from mistakes has led to improvements in the state-of-the-art, with the promise of more to come, since today's techniques are far from perfect and the opportunity for making mistakes is virtually inexhaustible. Therefore, the design and development of today's aircraft engines, as well as those of tomorrow, are derivatives of a "lessons learned" approach spread across a wide range of disciplines.

To understand this approach, it is helpful to first look at the "Big Four" requirements in aircraft engine design:

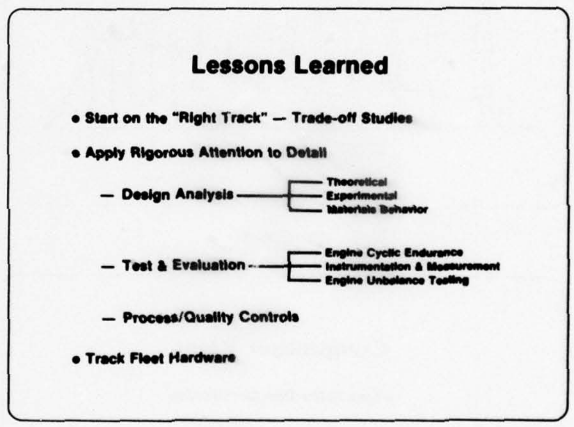
- performance (including emissions, noise, etc.)
- reliability—durability—maintenance
- weight
- initial cost—timing

An examination of these shows that performance is usually known shortly after the first engine goes to test. If the performance falls short of expectations, major improvement programs are launched immediately to both understand and correct deficiencies. The weight of the engine is known even before the drawings go to manufacturing, and certainly before the first test. The initial cost is reasonably understood and generally predictable based on the learning curve approach and our considerable manufacturing expertise and experience. The timing is also known as the parts are being manufactured, since "cost and delivery" data are closely coupled and generally interdependent.

However, in total contrast, mechanical reliability, durability and maintainability are relatively, and sometimes completely, unknown for a considerable time, often extending beyond certification. By this time, production tooling has been committed and fleets of hardware are in use. Mechanical failures, resulting from oversights and unknowns in the complex engine structures, are by far the most dominant and detrimental influences. Only by applying combined analytical, experimental and developmental techniques and tools can the impact of intermediate and long term, unexpected mechanical failures be minimized.

Short term or early mechanical failures during the engine test program are to be considered a blessing since, like performance deficiencies, these can be dealt with immediately. As someone once aptly said, "The worst thing we can do is make the parts work 98% of the time". This often incurs extensive and costly retrofit programs after many parts have been built, shipped and placed in service.

Having defined mechanical failures as the most critical factor affecting reliability, durability and maintainability, let us take a look at the key lessons learned on how to either eliminate or minimize these failures.



Start on the Right Track

Trade-off Studies

First of all, we need to start on the "right track" by making the proper trade-off studies relative to the design configuration and how the engine will be used to meet the aircraft system requirements. When a design starts out on the "wrong foot", it is difficult to fix and usually ends up being "patched" and "crutched" to meet requirements. Too often, the temptation is to live with these initial configuration decisions by providing field support, although it may mean something less than complete satisfaction with the product. Since engineering involves art, science and business, we must be sure in the beginning that all the science and business aspects are applied to thoroughly evaluate the art before the design is committed. Experience has shown that good trade-off studies are the key to making sure the design starts out on the "right track". Let us take a look at a few examples on how trade-off studies have helped the design approach.

Figure I shows a compressor rotor spool consisting of a "ring-drum" construction to minimize numerous bolted flanges and separate rotor parts. With this design, a more rigid, no-shift structure is formed by the integral disk-spacer configuration. The blades in the integral disk-spacer spool have circumferential dovetails to accommodate assembly and replacement without spool disassembly. The blades are assembled into the circumferential track through a loading slot; then all blades are shifted circumferentially so that the locking block retainers can be radially shifted into the lock slots. The design is basically a simple, statically determinate structure that minimizes parts, reduces cost and weight and, at the same time, can be readily analyzed to assure high reliability and durability while providing exceptional maintainability features.

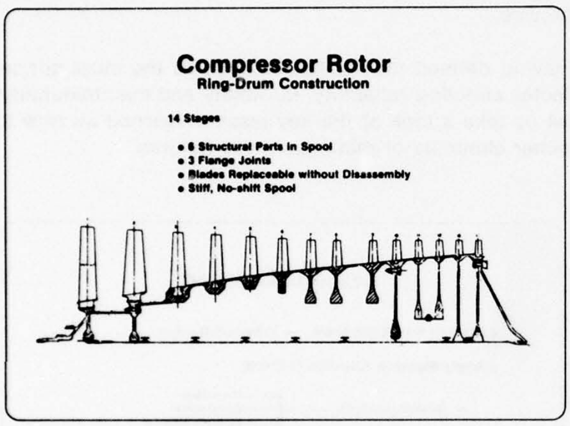


Figure 1

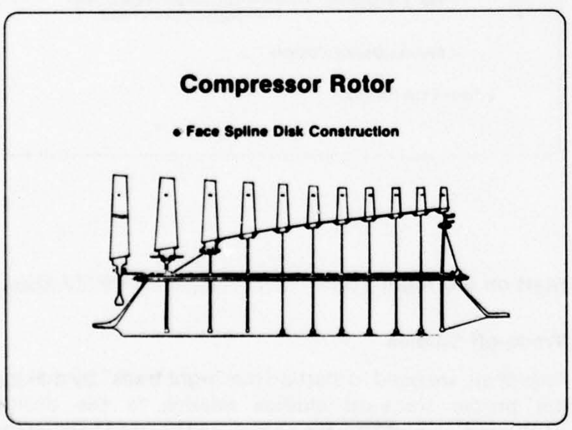


Figure 2

Let us compare these trade-off results with the eleven stage compressor from which it was originally evolved. Figure 2 shows this earlier engine compressor rotor featuring face spline disk construction, tie bolts, and separate rim spacers to form the flowpath. Note the multiplicity of parts in the earlier design and that the blades could not be replaced without disassembly of the rotor spool. The rotor in Figure 1 has only six structural spool parts for fourteen stages, while the earlier eleven stage rotor (Figure 2) has twenty-three structural parts and fourteen joints. Numerous contact points between mating parts produced considerable fretting and potential failure initiations. The substantial improvement in design configuration of the newer "ring-drum" rotor construction (Figure 1) compared to the face spline disk design (Figure 2) was the result of extensive trade-off studies involving life, weight, cost and maintainability.

Another example of the value of detailed trade-off studies is the high pressure turbine disk shown in Figure 3.

Figure 3

This design eliminates bolt holes through the highly stressed disk web where the Poisson ratio effect in a biaxial state of stress produces a reduction in axial thickness. When through bolts are used, this reduction in disk web thickness, although relatively small, results in a significant decrease in axial clamping load. Rim bolts to clamp the blade retainers are put through the dovetail rim posts where this stress effect is considerably less. Short spacers integral to the disk include the flanges with lower stresses, thereby permitting high bolt solidity and increased clamping. This feature minimizes shifting of parts in rotating assemblies which has caused rotor unbalance and fretting that leads to low cycle fatigue failures.

These examples are typical of actions that need to be taken in combination with proper material selection to assure "starting on the right track".

Apply Rigorous Attention to Detail

Design Analysis

☐ Theoretical— Once we have made the trade-offs and the design is started, hopefully on the right track, we need analyses capable of providing accurate states of stress caused by temperature effects, centrifugal forces and external pressure loading. Figure 4 illustrates a disk post failure resulting from an underestimated stress concentration. It highlights the need for a more rigorous design analysis in the beginning.

Figure 5 shows a finite element model of the dovetail rim post for the disk shown in Figure 3.

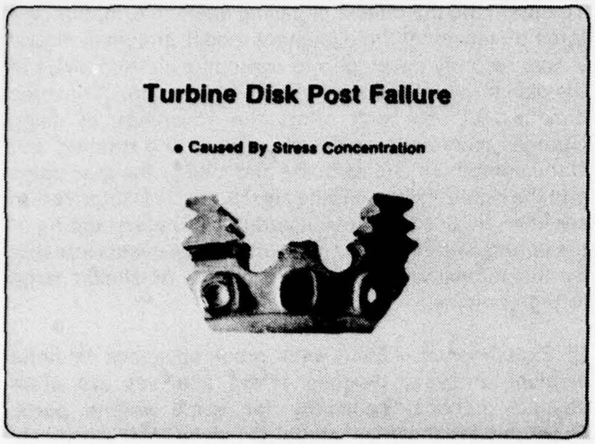


Figure 4

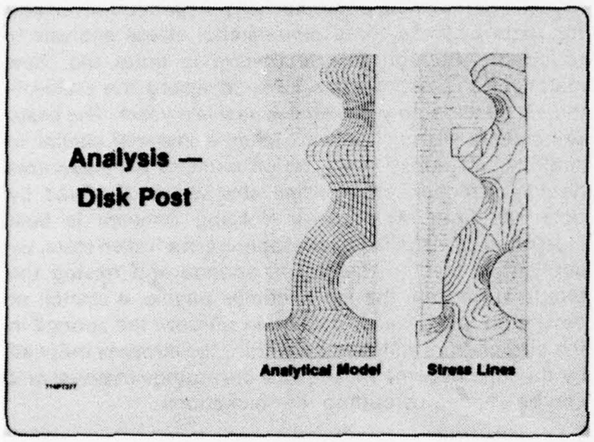


Figure 5

This analytical model is used to determine and understand the effect of stress concentrations and thermal gradients on the state of stress necessary to avoid subsequent mechanical failures. The resultant computer output, in terms of the stress gradients in the rim post, is also shown in Figure 5. High stress concentration regions are easily recognized by the grouping or concentration of the lines of constant stress. The bolt holes in the rim post, the base of the dovetail adjacent to the pressure surfaces, and the transition of the post to the hoop stress-carrying portion of the disk rim are all regions of high stress intensities. Note that the bolt hole in the analytical model of the rim post shown is circular but the disk (Figure 3) was made with a shaped hole. This tear-drop shaped hole was modeled and adopted after it was found to have a lower rim post stress concentration than the circular hole. Detailed stress analysis models made practical by sophisticated computer programs provide an effective method for designing durable, long life parts.

Finite element analyses can be applied to overall structures, such as the drum rotor shown in Figure 1 and modeled in Figure 6.

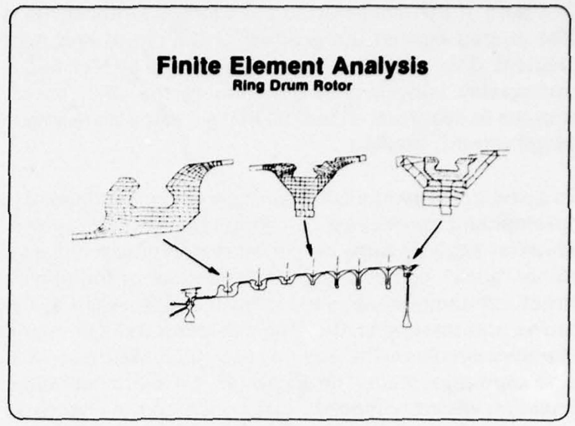


Figure 6

In this case, the finite elements are shells accurately defined directly from the numerical solution of classical shell differential equations. The shell stresses in the spacers due to centrifugal loading and meridional bending at the discontinuities of the disk-spacer connections, can be accurately determined and modified. The resulting deflections are used to set both radial and axial clearances for blades and vanes. Before making substantial investments in tooling to build the parts, analyses must also be made to determine the feasibility of extremes in design configurations and the susceptibility to premature failures.

Figure 7 shows an axial fan or compressor blade with a high twist gradient from root to tip. This twist gradient is a function of the blade tip speed and radius ratio (hub radius/tip radius). Blades with higher tip speeds and lower radius ratios have higher twist gradients. An examination of the steady state, chordwise stress distribution for both the convex and concave airfoil surfaces shows a considerable variation from the average stress. Care must be taken to avoid excessively high "end effect" stresses at the blade root. These can significantly reduce the capability of the blade to absorb vibratory stresses and may result in a high failure rate. The deflected shape of the blade under dynamic conditions must also be determined by analysis to correlate with aerodynamic and aeromechanical requirements.

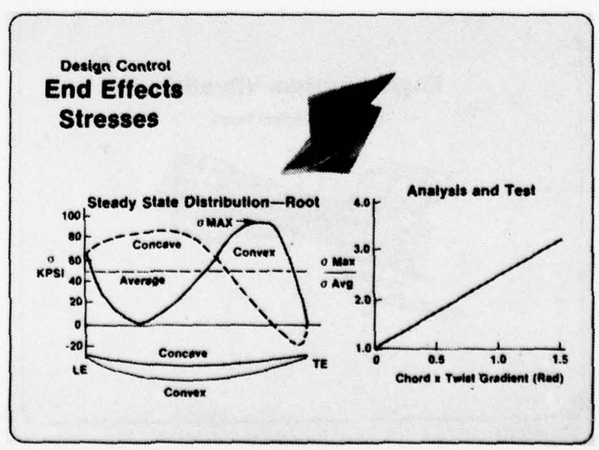


Figure 7

The ratio of the maximum to the average airfoil stress is also plotted against the *product of the chord and twist* gradient. This useful tool has been verified by test, and is particularily important in determining the "end effect" stresses to avoid loss of fatigue margin in low aspect ratio (length/chord) blading.

To avoid fundamental configuration errors and minimize development problems, a complete engine system vibration analysis must be conducted. Figure 8 shows a planar "stick" or beam and spring model of the engine structural components simulating the stiffnesses of the casing and rotating shafts. The assumed axi-symmetric structure simulates the engine reasonably well except at the mounting system. The exception is a minor limitation, since the vibratory modes most critical to the engine do not involve the mounting connections. Model simulation of the engine structure has been successfully used to predict vibratory modes and responses to unbalance and dynamic loading, including circular whirl and non-linear effects. Measurements of the dynamic response of the engine with accelerometers and strain gages mounted on the bearing housings have shown this planar model to be an excellent representation. Loads transmitted to the airframe through the pylon support structures have also been calculated and show agreement with measurements.

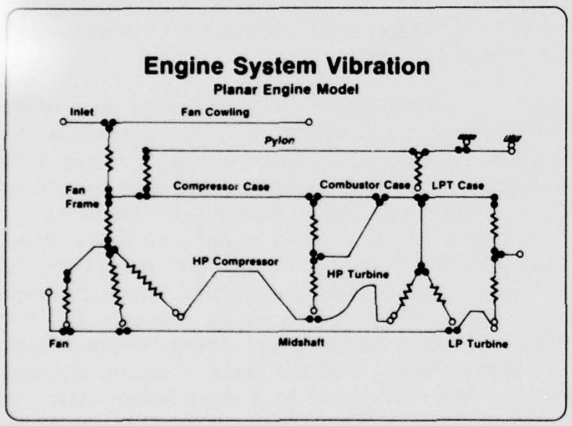


Figure 8

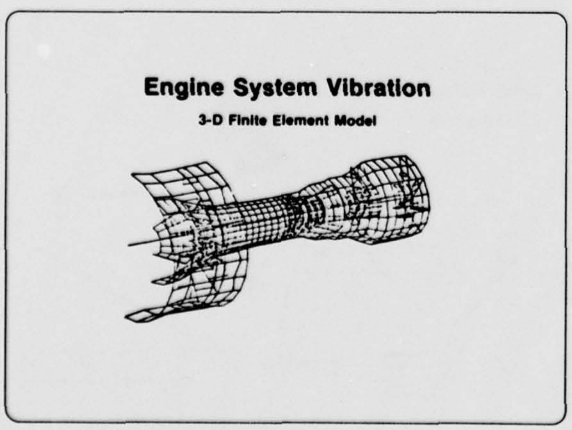


Figure 9

To determine the effects of casing shell deformation, the three dimensional finite element model, shown in Figure 9, was recently developed to overcome shortcomings of the planar (beam and spring) model simulation. The three dimensional approach offers the advantage of using changes in radial clearances between the rotating and static components to evaluate and modify the load paths and the rigidity of the engine structures. This information can then be used to provide additional understanding of the engine system during high unbalance events initiated by the ingestion of birds, tire treads, or similar large foreign objects.

☐ Experimental— Even with great advances in finite element analysis, detailed stress analyses are often beyond practical capability for some engine parts. Experimental analytical methods can be used effectively to determine the stresses in parts that are difficult to model. For instance, photoelasticity is particularily useful for parts where a three dimensional stress analysis is required. Although this technique is quite old, new materials and approaches have advanced the state-of-the-art considerably during the past few years. The basic principle is stress "freezing" using a material similar to Araldite, an epoxy resin which exhibits a measurable double-refraction effect when stressed and viewed by polarized light. The stress freezing concept is best illustrated by visualizing a wet sponge in a frozen state. By applying a load to the frozen sponge and raising the temperature until the ice becomes pliable, a stretch or deflection takes place. If we then refreeze the sponge in the stretched or deflected position, the stresses induced by the load become locked into the sponge material and can be studied by cutting it into sections.

Figure 10 shows an Araldite model of a bevel gear drive used to determine the three dimensional stresses in the gear teeth. The gears are mounted in the engine housing which serves as a loading fixture. A scaled torque is applied to the gear train; then the applied stresses are locked in and subsequently determined by sectioning as illustrated.

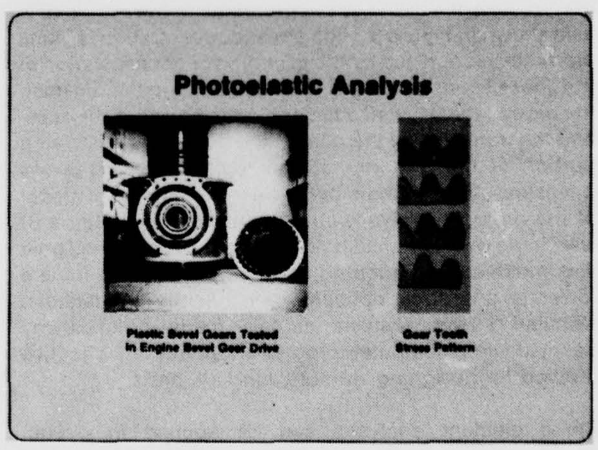


Figure 10

Figure 11 shows a photoelastic Araldite model of a pin root blade with a scaled radial load applied to simulate a centrifugal force at the root. This model was used to evaluate the stress concentrations at the hole locations where cracking was observed in the aft tang. The intensity of the fringe patterns indicates the presence of large stress gradients at the sides of the hole.

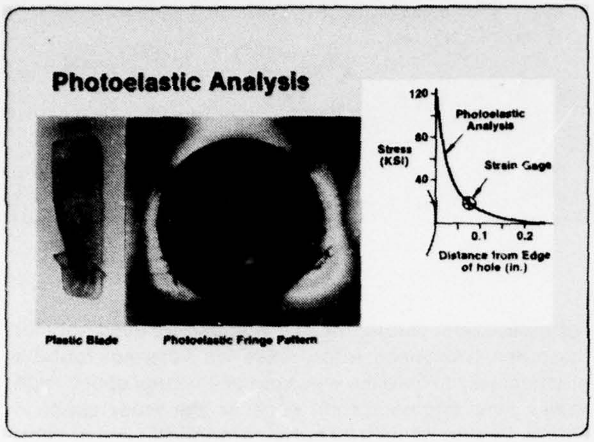


Figure 11

The stress level at the blade root hole is plotted as a function of the axial distance away from the hole surface. A data point obtained by a strain gage placed on the blade root and engine tested at full speed indicated a stress of 20,000 psi at .080 inches from the hole edge at the three o'clock position. Since the gages are finite in size and a retaining pin is used in the hole, strain gages could not be placed closer to the edge. Note that the stress gradient as analyzed by photoelastic methods agrees with the engine strain gage value, but continues to rise rapidly to the surface of the hole where the stress is six times higher than at the engine gage. It is also interesting to note from the fringe pattern in Figure 11 that a still higher stress exists at the four o'clock position and relatively far from where the engine strain gage was located. In addition to stress prediction capability, these techniques are particularily helpful in determining the proper location for engine strain gages.

☐ Materials Behavior— Understanding all aspects of materials behavior under engine operating conditions is essential to a successful design and development process. The standard mechanical properties relating to static strength, rupture and fatigue are utilized at the outset of preliminary design. However, more subtle time-dependent effects involving creep-fatigue interaction, load strain cycling and quality considerations have a major impact on parts life. As a result, these material characteristics must be integrated into the stress and life analysis of the parts before final release to prevent costly surprises.

Test and Evaluation

The objective of the test and evaluation program is to experimentally evaluate the product by obtaining stress, temperature and vibration data in order to develop and improve the design.

☐ Engine Cyclic Endurance— Several testing techniques have been developed over the years, but none has achieved more success, or wider acceptance, than the cyclic endurance test. Figure 12 illustrates an accelerated endurance cycle simulating a typical subsonic transport aircraft mission. The abbreviated mission cycle was developed because it is impractical to include the long climb, cruise and descent legs of the flight cycle, which may be from one to four hours in length, depending on the aircraft and mission. Substantially shortening the actual flight cycle, but retaining appropriate engine speed and temperature conditions, provides sufficient cyclic endurance operation to simulate aircraft missions in time to "lead the fleet". The goal or objective is to have a factory test engine accumulate more pressure, speed and temperature cycles than any single operational engine in service. This approach identifies mechanical failure modes before they occur in operation and with sufficient lead time to permit timely corrective action.

During the takeoff and climb portion of the cycle, the engine is run to high speed and power producing the maximum temperature and stresses. The cruise, throttle retard to idle, accel to reverse thrust power, and shutdown portions of the cycle produce stress-strain reversals due to load, temperature and thermal gradients simulating those in service.

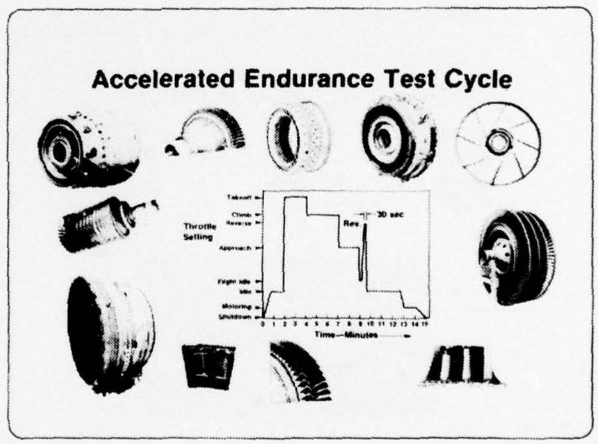


Figure 12

By using the fleet leader cyclic endurance concept, numerous design limitations and deficiencies have been uncovered for engine components similar to those shown in Figure 12. However, not all parts are subjected to the maximum stress-strain range produced by actual flight cycles because the long-time creep effects and maximum thermal stress reversals caused by cooling are not completely simulated. Internal turbomachinery parts, such as disks, take considerable time to heat and cool. This is due to their relatively high mass as compared to the lower mass flowpath parts such as the combustor, sheet metal parts and airfoils. As a result, the larger mass parts tend to stabilize at intermediate temperatures during the short endurance cycle instead of experiencing wide ranges of unequal heating and cooling which produce high thermal stresses in actual flight cycles.

Because of these limitations, accelerated cyclic testing alone cannot screen out all deficiencies for all components. To be sure, we need to carefully determine the stresses, temperatures, thermal gradients and vibration characteristics from component and engine tests. This information must be used in the stress and life analyses of components with particular emphasis and attention to those that do not respond to cyclic endurance testing.

□ Instrumentation and Measurement— To record stress, vibration, temperature and pressure parameters, the engine must be adequately instrumented with sensitive equipment. Thermocouples and strain gages, as shown in Figure 13, are used to record both steady state and transient temperatures and stresses during engine operation. Considerable rework of hardware is often necessary to accommodate instrumentation lead wires and tubes. When instrumentation is applied to rotating parts, a slip ring mechanism must be attached to the shaft to bridge the lead wires on the rotor to the stationary leads. Over the years, this rotating component has been a weak link in the measurement system. The number of channel readouts are limited; noise is generated on the brushes causing signal interference; air cooling is required and shaft vibration causes rapid breakdown and premature maintenance. Nevertheless, significant progress has been realized with high speed slip rings over the past twenty five years.

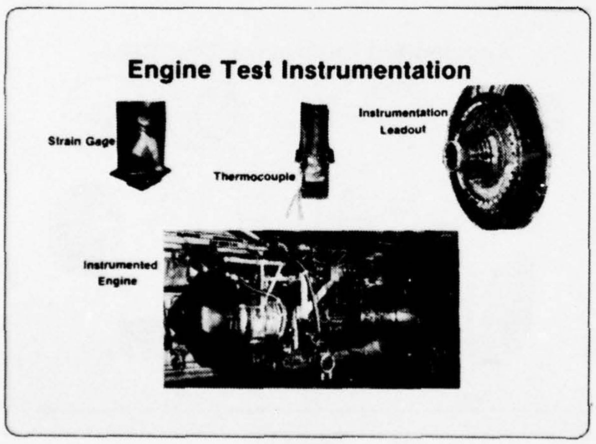


Figure 13

Figure 14 shows a comparison between a standard current technology slip ring and an improved type developed at General Electric. The new, miniaturized slip ring is smaller, lighter and less costly but, more importantly, it can operate at higher rotational speeds with longer life, higher quality readout and with 60% more gage capacity.

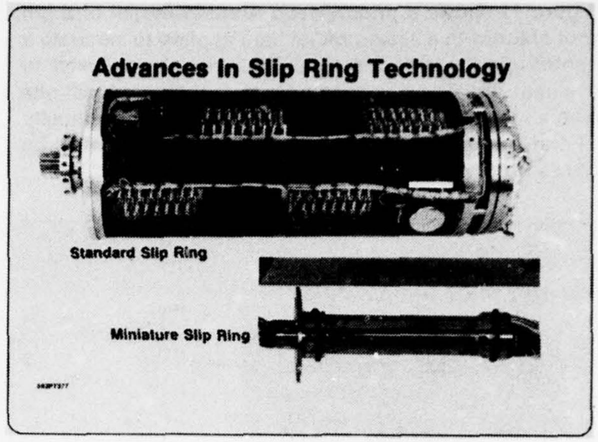


Figure 14

For multi-spool rotors, the slip ring can be used only for the outer, low speed rotor, since we have not found a practical way to lead the wires out of the inner spool. High speed core engine rotors, such as the inner spool in Figure 15, can be instrumented through the use of short range radio-telemetry where radio waves replace lead out wires. This is shown schematically in Figure 16. The instrumentation leads are connected to a transmitter-antenna system where the sensitive signals are transmitted by radio waves to a receiving antenna. The signals are then read out on an oscilloscope in the same manner as with the direct wired slip ring. An application of a radio-telemetry transmitter is shown in Figure 17. Relative to slip rings, the number of read out channels is limited, but it permits data acquisition under correct environmental conditions of the high pressure rotor system.

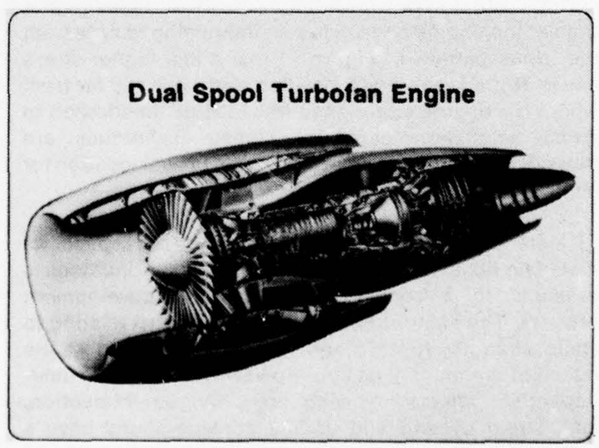


Figure 15

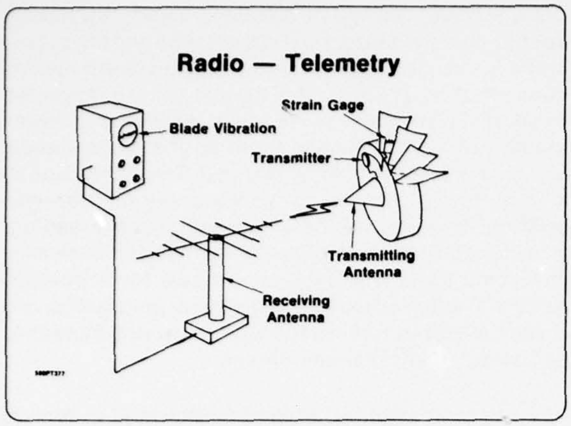


Figure 16

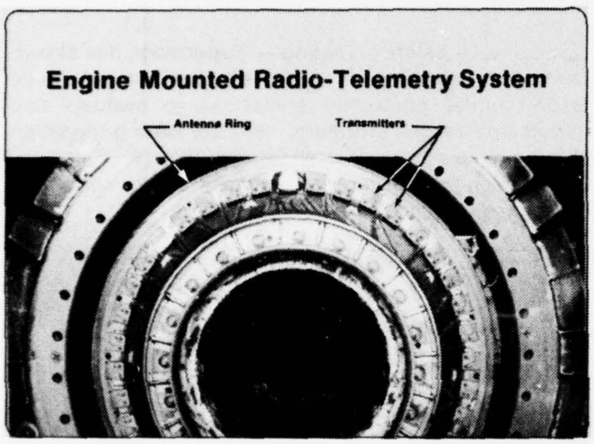


Figure 17

Hot parts, such as turbine blade airfoils, introduce additional complications for instrumentation. These parts can operate at metal temperatures in the range of 1800° F, gas temperatures of 2500° F and with centrifugal forces of 80,000 g's . The lives of strain gages, thermocouples and lead out wires in this environment are extremely limited, if not impractical, using routine techniques.

The infrared optical pyrometer, as shown in Figure 18, is a light sensitive device that can be used to measure the temperature of rotating turbine blades from a stationary position. The optical device is calibrated relative to a known temperature source, preferably a reference thermocouple imbedded in the lower temperature and lower stress region of the airfoil to maximize its readout life. The pyrometer measures both the absolute temperature level and thermal differences between airfoils. When wired into the engine control circuit, it can be used to control engine fuel flow instead of the traditional exhaust gas thermocouple harness. Since metal temperatures are measured directly, there is less chance to over-temperature the turbine blades during an acceleration to full power.

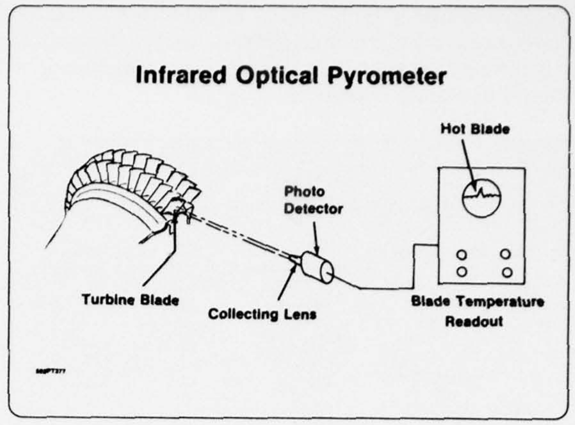


Figure 18

The same pyrometer is also an extremely useful tool for evaluating the design effectiveness of high temperature, turbine blade, cooling systems. Figure 19 shows two different cooling configurations, assembled into the same turbine rotor and subjected to accelerated endurance cycles. Design B has fewer but larger holes at the leading edge and one row of shaped holes on the forward suction surface. Design A, with a greater number of smaller holes at a higher angle to the surface, uses slightly more cooling air than B. The pyrometer trace measuring blade metal temperature shows a smooth even temperature trace at 1800° F for Design A blades with the more evenly distributed film cooling; whereas, the temperature profile for Design B is erratic and also more than 100° F hotter. This useful evaluation tool has been extremely valuable in accelerating the design and development cycle of high temperature, air cooled blading for both military and commercial engines.

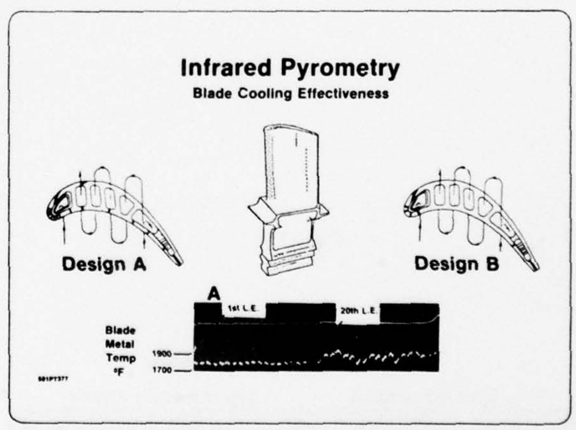


Figure 19

The radial and axial positions of the internal rotor blades, shrouds and seals affect aerothermodynamic performance and internal rotor thrust balance which, in turn, controls the bearing axial loading. As a result, it is often desirable to determine the radial and axial positions of the rotating blades and seals relative to the stator during normal and transient engine operation.

Figure 20 shows a high energy (eight million electron volts) X-ray system. It is used to "see" through the engine to determine internal radial and axial clearances at different operating conditions.

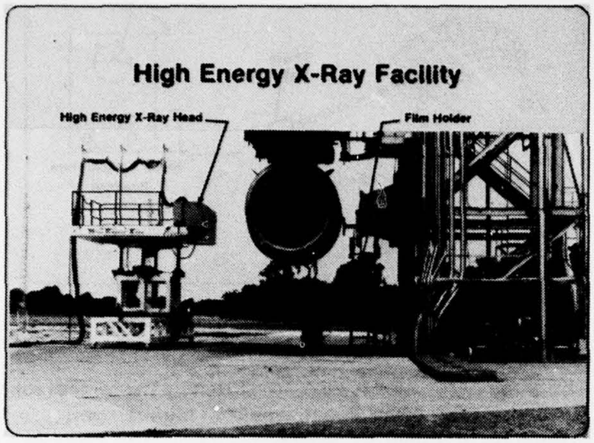


Figure 20

The low pressure turbine shown in Figure 21 was analyzed by use of high energy X-ray images at several engine power levels. Resulting X-rays of the seal-tooth area at two conditions are shown in Figure 22.

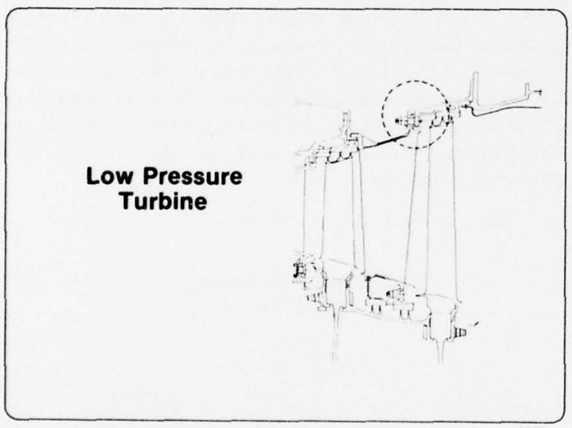


Figure 21

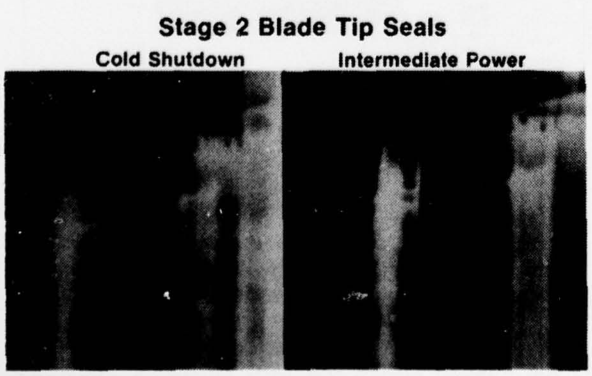


Figure 22

During a "cold" shutdown transient, where the turbine rotor has not reached thermal stabilization and the engine throttle is retarded, the first tooth of the blade shroud has moved aft, almost to the end of the stationary honeycomb shroud seal. The honeycomb in the picture is of lower contrast relative to the solid tooth on the blade shroud because it is of much thinner material. The aft position of the seal tooth was not a concern during transient operation. However, at the cruise performance setting, the blade seal tooth is actually riding off the aft stationary honeycomb seal. This extreme, aft seal tooth position results in a high leakage rate at the blade tip resulting in a substantial loss in turbine efficiency requiring increased fuel flow to maintain engine thrust.

The X-ray technique has proved to be useful in solving engine deflection and clearance problems not readily afforded by other methods. It is to be noted that the radial clearances can be more accurately measured from the X-rays when the honeycomb depth is measured at assembly.

☐ Engine Unbalance Testing— Experience has shown the need for moderate to high levels of engine unbalance testing under controlled conditions to evaluate and understand engine structural behavior during ingestion of large birds and other large foreign objects. This need triggered the design and development of the controlled unbalance device shown in Figure 23.

Figure 23

The device consists of a radioactive depleted uranium weight mounted in the hub of the fan rotor and held in position by an explosively charged bolt. The weight is located slightly off of the engine center of rotation, and is constrained to move radially outward along a sliding track when the explosive charge is triggered by a signal transmitted through a slip ring. A lead cushion is placed at the end of the track to protect the disk bore from possible impact damage. RTV or silicone rubber is used to fill the cavity and act as a hydrostatic resistance to slow the weight and reduce the impact energy. A bolted-on cover plate is used to contain and totally enclose the sliding weight. The size of the uranium mass can be varied to provide a wide range of unbalance. This technique has been used successfully to simulate engine unbalance at full power, such as experienced with bird or other foreign object ingestion.

When very high levels of unbalance are required to simulate severe ingestion events, which occur approximately once in a million flight hours, fan blade airfoils can be broken off in a controlled manner, also by the use of explosive charges. Figure 24 shows a fan blade with an explosive charge bolted to the airfoil. When the charge is triggered by a signal transmitted through a slip ring, the airfoils are burned through. In one such test, this resulted in 70,000 gm-in. of unbalance per blade to simulate the ingestion and subsequent major engine damage.

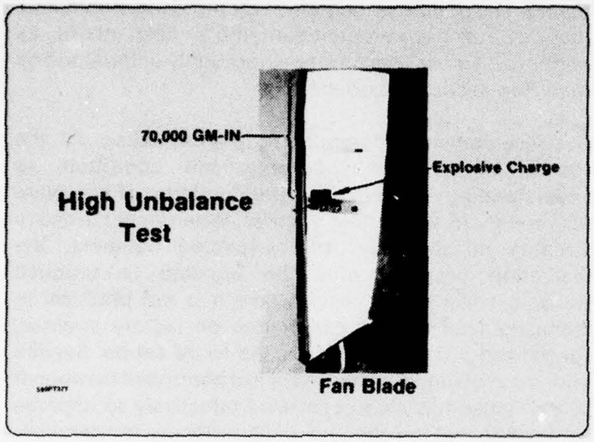


Figure 24

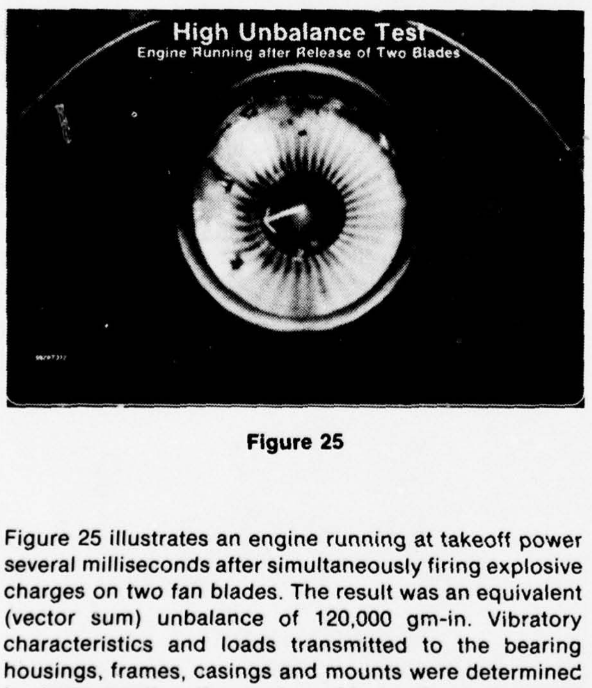


Figure 25

Figure 25 illustrates an engine running at takeoff power several milliseconds after simultaneously firing explosive charges on two fan blades. The result was an equivalent (vector sum) unbalance of 120,000 gm-in. Vibratory characteristics and loads transmitted to the bearing housings, frames, casings and mounts were determined by instrumenting the engine with accelerometers and strain gages. Although these high unbalance tests are destructive, much can be learned in advance of possibly experiencing similar events in service, and necessary modifications can be made before large numbers of fleet engines are committed to service.

Process/Quality Controls

The high degree of sophistication and complexity associated with the design of aircraft engines results in an increasing dependence on the manufacture of high quality components. As a result, significant effort has been applied toward improving the materials, processes, machining and inspection methods. Many lessons have been learned but few more fundamental than recognizing that quality is essentially dependent on a controlled process rather than inspection.

About fifteen years ago, the demand for higher strength materials with consistent mechanical properties was met with the introduction of vacuum melting technology. This advance in processing also permitted the introduction of a new series of harder to forge "superalloys" with many manufacturing challenges. Realizing the difficulty in relying upon Non-Destructive Testing (NDT) to provide inspection methods for detecting and eliminating low quality parts, attention was shifted toward improving the manufacturing process.

Upgrading the quality of these new alloys was accomplished with the introduction of Premium Quality (PQ) material, a new concept providing a disciplined approach in the processing of raw materials. This careful selection involves screening and sampling of basic alloy ingredients, followed by careful monitoring of melting parameters during all phases of the vacuum melt sequence.

Attention to details, such as avoiding solidification conditions that produce segregation, routinely clearing the melt furnaces to minimize contamination, and cleaning and inspecting ingots prior to the final melts are now standard practices. Further control is applied during the ingot-to-billet conversion to maintain melt traceability and ingot position. Special Non-Destructive Evaluation (NDE) monitoring is used on the final billet and indications are often cut out for metallurgical evaluation. Depending on the frequency and distribution of the indications, a complete ingot or entire heat of material can be rejected. Certain components, such as billets for rotating disks, undergo further NDE at the forging source. Acceptable billets are cut into forging multiples (one part per multiple), and etched on both faces prior to forging as a final check for possible segregation. In the event of unacceptable indications, correlation is made to the location of the questionable billet in the ingot and, depending on the acceptance criteria, the ingot is again subject to rejection.

Forging and heat treat operations are performed to very detailed practices with tight controls on forging pre-heat and reduction schedules and the quench rates after solution heat-treatment. Metallurgical control is maintained over morphology, grain size and mechanical properties. Once accepted as Premium Quality material, forgings are shipped to the shop and skim-cut to a configuration suitable for immersion ultrasonic inspection. A three-mode scan is performed with Numerically Controlled (NC) equipment capable of finding randomly oriented indications in the part. Rejections are less than one part in one thousand for significant ultrasonic indications, and few of these have proved to be actual flaws. This attention to detail relative

to process controls by both the supplier and manufacturer provides assurance that high quality parts are produced. Cases where part failures were attributed to quality problems have usually been traced to a "breakdown" in the discipline of the process control sequence.

In addition to the process control procedures developed to produce critical engine components, there are specific actions that should be taken to increase the probability of manufacturing successful parts. For example, disciplined release of new processes is essential to avoid both short and long range problems. Pilot or small scale process development prior to a full scale production release often identifies potential problems, and also leads to better understanding and definition of the technical approach used to produce and accept hardware.

Avoiding "risk release" items is another way to minimize trouble. Items in this category are those where the process is not completely defined or developed, but where a release is made on a "best efforts" approach. This type of release should be confined to a development program.

Sometimes, a part is designed and dimensioned with a very wide tolerance range on critical thicknesses and radii that are relatively inaccessible. Experience has shown that these "liberally dimensioned" parts are often subject to high failure rates. Action must be taken to provide close dimensional control in the inaccessible areas. This requires proper tolerancing of drawings so that proper processing and inspection controls can be put in place.

The need to use proven vendors and suppliers has also been demonstrated as a way to avoid unexpected problems in both the development and production programs. The "buy in" concept, where a vendor with inadequate technical capability bids low to get an order, usually results in a high cost situation. Usually, there are no "short cuts" worth taking in the delivery of quality hardware. Periodic checks, by cut-up of hardware for inspection of dimensional and metallurgical properties, must be made in-house, as well as at the vendors' plants. This precaution helps avoid future call-back of defective parts.

Finally, two other important, but often overlooked, aspects of manufacturing high quality hardware are the need for a simple documentation system and a manageable process change approval system. The importance of having the entire manufacturing process well documented becomes clear when unacceptable hardware is discovered. Having a well documented history of the processing from the "mine to the assembly area" becomes extremely helpful in determining the extent of corrective action required. A parallel and supporting effort that eliminates "red tape" in obtaining timely approval to process changes is also necessary. Surprises, in terms of reduced parts life, have occurred because the vendor made "unauthorized" changes in the process to meet the pressure of schedules, after unsuccessfully encountering "red tape" in getting a change through the system. Dealing with vendors in a completely straight forward manner, with open communication channels, will minimize mistakes and is fundamental to the success of the overall process/quality control plan.

Tracking the Fleet Hardware

When engines are deployed in fleet service, it is essential to track and understand the performance, operability and life relative to initial requirements. The condition of fleet hardware needs to be continually examined for possible deviations from factory experience and requires careful analysis. It is not uncommon to find problems occurring in fleet hardware that were not experienced in factory testing. The effects of long time creep and more complete cool-down of hot parts encountered in fleet service, as mentioned earlier, can be more thoroughly understood by analyzing service experience.

The development of computer programs dealing with the effects of operational severity can contribute to understanding differences in the condition of hardware between factory and fleet engines. When fleet hardware deviates considerably from expected behavior, the evaluation program must be updated to produce equivalent results. In cases where it is not practical to duplicate fleet service experience on factory engines, specialized component tests need to be set up. Service evaluation of modified hardware in a controlled number of fleet engines has also been used effectively to improve confidence that the changes will be effective in operation.

Integrating the Lessons Learned

Undetected failure modes must be extremely elusive to "run the gauntlet" of configuration trade-off studies, design analyses, evaluation techniques and numerous process/quality control measures typified in this paper. Nevertheless, it can and does happen; hence, extreme care must be taken to assure disciplined application of check lists derived from the many "lessons learned" over these past twenty five years. When rigorously applied in an integrated, orderly manner, we can hope to avoid these more difficult to detect failure modes.

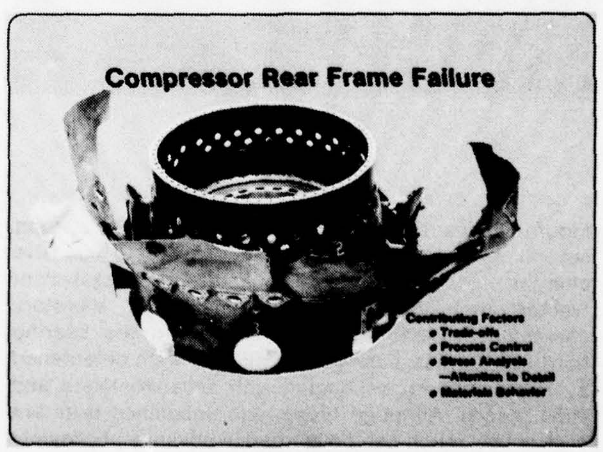


Figure 26

Figure 26 illustrates an example of a configuration failure involving complex interactions between the design, design analysis, materials behavior, manufacturing and process/quality controls. The engine compressor rear

frame structure shown is essentially a fabricated pressure vessel subjected to repeated thermal and pressure loading caused by engine start-stop cycles and throttle movements. Imperfections in difficult-to-inspect weldments led to several failures of this configuration during factory cyclic endurance testing. A detailed investigation revealed weld imperfections, substantial local stress amplification due to weld mismatch and, more importantly, new information on crack growth behavior leading to a significant advancement in the understanding of fracture mechanics and life prediction. The failure of this fabricated structure led to the identification of the potential cyclic life reductions resulting from combined processing and materials behavior characteristics shown in Table I.

Potential Cyclic Life Reduction
(INCO 718 Weldments at 1000°F)

Anomaly	Fatigue Strength Reduction from 10,000 Cycle Life	Stress Concentration Factor
Plate Misalignment (Production Quality Range)	30% — 80%	1.2 — 2.1
Weld Bead Contour (Production Quality Range)		1.3 — 3.5
Weld Quality		
Porosity	10%	1.26
Low, Undercut or Repair Weld	16%	1.2 — 1.55
* Lack of Fusion	57%	5.0
* Lack of Penetration	81%	5.0
Inherent Flaw Size	Up to 50%	—
(.040 — .120)		

* Outside of Production Limits

Table I.

Analytical techniques and approaches used in the design analysis of pressure loaded, fabricated, shell structures were modified to include these findings. In addition, improvements were put in place relative to design and drafting practices, manufacturing tooling, and process/quality controls.

Mechanical failures are costly and humbling experiences. However, when these failures are induced early in the development program, the lessons learned more than offset the cost, provided they are new lessons and not just a repeat of ones already experienced. By continued application and building upon these experiences, we can project continued progress in improving reliability, durability and maintainability for the future.

SOPHISTICATION ET FIABILITE

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RESUME

Indépendamment du compromis entre la sophistication et les performances qu'il se contente d'évoquer, l'auteur montre que lorsque la sophistication est liée à des problèmes de fiabilité, le compromis à faire réside essentiellement entre les divers aspects de la fiabilité d'un moteur (aspect sécurité, aspect opérationnel, aspect économique).

Il attire l'attention sur les différences qu'il y a sous cet angle entre les parties structurales d'un propulseur d'avion et ses systèmes de régulation et de contrôle.

Par quelques exemples, il montre notamment les différences de conception que peuvent imposer des utilisations sur avions monomoteurs ou avions multimoteurs.

En conclusion, tout en reconnaissant les vertus et la nécessité des techniques redondantes, l'auteur attire l'attention sur leur côût et leur difficulté de mise en oeuvre.

AVANT-PROPOS

Il est d'usage dans la presse et les manifestations aéronautiques de s'extasier devant la sophistication et la fiabilité des matériels modernes, d'autant que ces caractéristiques apparaissent souvent contradictoires.

L'enthousiasme des utilisateurs et des constructeurs est plus nuancé car ils en connaissent le prix, les premiers en coût d'achat et d'exploitation, les seconds en efforts de recherches et d'essais pour tenter de concilier ce qui semble inconciliable.

Nous allons tenter d'identifier où se situe exactement le problème et de voir, lorsqu'il y a antagonisme entre la sophistication et la fiabilité, dans quelles voies il faut chercher le meilleur compromis.

QU'ENTEND-ON PAR SOPHISTICATION ?

Un matériel dont la fonction première reste inchangée devient, à mesure que progresse la technique, de plus en plus compliqué ; on utilise pour l'élaborer des matériaux et des procédés de fabrication de plus en plus coûteux : au sens que l'on donne à ce mot en technique actuellement -et il s'agit là probablement d'une légère impropriété de langage- il devient de plus en plus sophistiqué. Est-ce normal et pourquoi ? Telle est la question que nous devons nous poser.

En fait, outre la propriété de base du matériel (pour un moteur d'avion par exemple le niveau de poussée) on exige de voir améliorer d'autres propriétés intéressant soit les performances (niveau de bruit, consommation spécifique), soit la sécurité (minimiser sinon supprimer les risques de mettre l'avion en danger), soit le coût d'exploitation (diminuer le nombre et le coût des interventions de maintenances).

Notre propos ne sera pas d'analyser en détail les compromis qui peuvent avoir amené à définir les réglements de sécurité auxquels doit se conformer le constructeur ou les performances imposées par l'utilisateur. Prenant ces compromis comme données de base, nous en examinerons les conséquences et les choix qu'il reste au motoriste pour délivrer un matériel aux coûts d'achat et d'exploitation minimaux. Il nous paraît toutefois important, dans ce cadre de rappeler que la notion de sécurité peut varier en fonction de l'utilisation. Sur un avion commercial multimoteurs, les réglements imposent la possibilité de décoller et de voler avec un moteur en panne : dans ce cas l'extinction moteur, à condition qu'elle ne soit pas trop fréquente, ne met pas en cause la sécurité de l'avion -de par la conception de celui-ci- et les seules pannes à prendre en considération pour la sécurité sont par exemple le feu, la contamination de l'air cabine ou l'éjection de débris non contenus.

Il en ira tout autrement pour un intercepteur monomoteur où la panne du moteur entraîne la perte sinon du pilote tout au moins celle de l'avion. On comprendra donc pourquoi, à la demande des utilisateurs, les systèmes de régulation des propulseurs des avions monomoteurs sont beaucoup plus sophistiqués que ceux des propulseurs civils.

LES DIVERS ASPECTS DE LA FIABILITE

Après avoir défini ce que nous entendons par sophistication du matériel, il nous paraît utile de rappeler la définition de la fiabilité : c'est la probabilité qu'a un matériel de remplir sa fonction dans des conditions déterminées et pendant un temps donné. La difficulté majeure est qu'un matériel a toujours plusieurs fonctions, plus ou moins indépendantes et la loi bien connue de l'ennui maximum veut qu'en améliorant la fiabilité de l'une vous ayiez toutes les chances de détériorer celle des autres.

Par exemple si pour assurer de façon certaine la fermeture d'un circuit carburant on met deux vannes en série au lieu d'une, on multiplie par deux la probabilité de non ouverture en cas de besoin. De même, assurer une bonne probabilité de coupure d'un moteur en cas d'urgence conduira à augmenter la probabilité de coupure intempestive en service normal.

Nous sommes donc amenés à différencier les fonctions et donc les pannes d'un matériel suivant qu'elles intéressent la sécurité, les performances ou plus prosaïquement l'aspect opérationnel de leur utilisation (retards, annulation de vol etc...) et le choix résidera essentiellement dans les fonctions à privilégier par rapport aux autres : en fait nous voyons donc que le compromis sophistication-fiabilité est surtout un compromis entre les fiabilités des diverses fonctions d'un matériel.

Pour un moteur d'avion, le problème se pose différemment suivant que l'on considère les parties structurales (rotor, stator et points de suspension) ou les équipements de contrôle et de régulation.

Pour les parties structurales, la redondance n'existe qu'au niveau des éléments d'assemblage ou de fixation interne du moteur. La plupart du temps une amélioration de la qualité matière ou du dessin se traduira par une augmentation du niveau de sécurité et simultanément par une augmentation de la longévité des pièces. Cette règle n'est pas absolue : il peut y avoir conflit, par exemple pour une aube fan entre l'endurance et la résistance au choc au niveau du choix des formes mais de tels cas sont suffisamment rares pour que nous puissions considérer qu'au niveau des structures, une amélioration intéresse l'ensemble des fonctions du matériel.

Pour les éléments de régulation et de contrôle, par contre, la redondance est souvent utilisée pour accroître le niveau de sécurité : dans ce cas, le doublement des chaînes, avec l'adjonction de dispositifs de commutation et de détection d'état introduit inéluctablement une multiplication des actions de maintenance corrective et même préventive s'il y a panne cachée.

Pour illustrer ces différentes considérations, nous vous proposerons deux exemples.

CE QU'IL FAUT EVITER

Le premier exemple est élémentaire et schématise ce qu'il ne faut pas faire en matière de sophistication inutile.

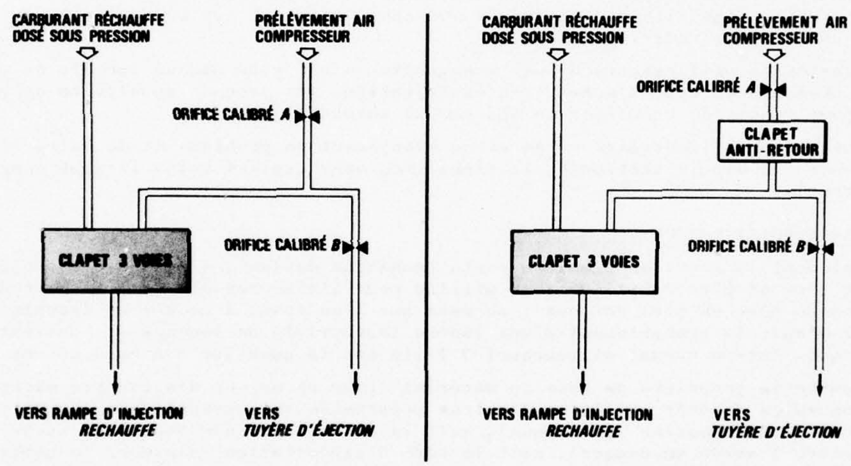

FIG. 1

SOPHISTICATION ET FIABILITE.– PURGE DE RAMPE RECHAUFFE

Un système d'alimentation en carburant de réchauffe est équipé d'un dispositif de purge de rampe pour les fonctionnements sans réchauffe pour éviter les risques de cokéfaction dans la rampe.

Le circuit est constitué d'un clapet 3 voies qui laisse passer le carburant vers la rampe s'il y a pression de carburant et qui laisse passer l'air sortie compresseur vers la rampe dans le cas contraire : cette alimentation en air se fait par un potentiomètre pneumatique constitué de deux orifices calibrés A et B ainsi que figuré en Figure 1.

La panne à craindre est la fuite de carburant dans le circuit d'alimentation de la purge, fuite dont une remontée vers le compresseur est considérée comme dangereuse.

Si l'orifice B n'est pas obturé, une fuite éventuelle sera drainée par cet orifice : la pression entre A et B étant inférieure à la pression sortie compresseur, il n'y a pas de risque de remontée de carburant vers le compresseur.

Certains esprits un peu puristes arguèrent du caractère dormant de la fuite du clapet et du colmatage du drain en B pour demander l'introduction d'un clapet anti-retour sur le circuit d'air pour "pallier toute éventualité" et furent malheureusement écoutés : le clapet introduit était malheureusement le moins fiable des éléments en présence et sa panne (coincé ouvert) était également dormante. Un rapide et simple calcul de probabilité tenant compte de temps éventuels

entre inspections montra que le niveau requis par la sécurité pouvait s'obtenir :

- soit en maintenant le clapet anti-retour en le vérifiant toutes les 100 heures
- soit en vérifiant le drain ou le clapet trois voies toutes les 3 500 heures ce qui permettait de supprimer le clapet anti-retour.

La conclusion est qu'il ne sert à rien d'introduire des sécurités dont il n'est pas possible de vérifier simplement l'état réel : le niveau de sécurité n'est pas nécessairement accru et ces matériels sont sources d'un alourdissement certain de la maintenance et de son coût.

Après cet exemple simple nous allons revenir sur le problème de la régulation des moteurs destinés à équiper des avions monomoteurs.

LE PRIX DE LA SECURITE SUR UN AVION MONOMOTEUR

Le choix d'un avion monomoteur entraîne pour l'utilisateur ou l'avionneur la nécessité d'imposer pour les arrêts moteurs sans possibilité de réallumer un taux beaucoup plus faible que celui accepté dans l'aviation commerciale où cette panne peut être considérée comme mineure. Il y aura donc sur un tel moteur ce que nous allons appeler un secours régulation et il va nous falloir définir quelles sont les pannes que devra pallier ce secours.

Sont exclues bien sûr de ce lot les pannes portant atteinte à l'intégrité du moteur et pour lesquelles il n'y a aucune parade.

Parmi les pannes justiciables d'un secours, il y a, par ordre de gravité décroissante :

- la rupture d'entraînement des accessoires moteur et avion
- la rupture d'entraînement des accessoires moteur, les accessoires avion étant toujours entraînés
- la rupture d'entraînement des accessoires moteur sauf de la pompe d'alimentation de réchauffe
- la rupture d'entraînement de la pompe du moteur sec, tous les autres accessoires moteurs étant toujours entraînés
- les blocages de manette de gaz ou de vanne de dosage
- les défaillances de la partie calcul hydromécanique
- les défaillances de la partie calcul électronique.

La fréquence de ces pannes est en général fonction décroissante de leur gravité. Aux deux extrêmes nous trouvons les pannes électroniques plus fréquentes mais assez facilement parables et la perte complète de l'entraînement d'accessoires, panne beaucoup plus grave mais heureusement beaucoup moins probable.

La panne électronique peut être palliée soit par redondance soit par le choix préalable de limiter l'action de l'électronique à des corrections des ordres d'un régulateur hydromécanique prépondérant : dans ce dernier cas, on assure, malgré une certaine dégradation des performances, la possibilité de ramener l'avion à sa base ou au terrain le plus proche mais l'électronique étant moins compliquée, elle aura moins de pannes qu'une électronique prépondérante et redondante et nécessitera donc moins d'interventions de maintenance corrective.

Quant à la perte d'entraînement des accessoires, elle conduira à prévoir un secours total incluant tous les éléments normaux d'une régulation : source de carburant sous pression, organes de dosage et organes de calcul du débit.

Entre ces deux cas extrêmes se situent toutes les solutions où l'on ne pallie qu'une partie des pannes possibles prenant le risque des autres en fonction de leur probabilité.

Dans tous les cas l'obtention d'une faible probabilité d'une extinction en vol aura pour contrepartie :

- la nécessité de vérifier périodiquement le bon fonctionnement du secours
- l'augmentation du nombre de missions annulées ou modifiées à cause des pannes du secours lui-même ou de pannes induites par le secours sur la régulation normale
- des incidences sur le poids et le prix qu'il n'est pas dans notre propos de développer ici mais qui peuvent être déterminantes.

CONCLUSION

En conclusion, nous pouvons dire que, surtout au niveau des systèmes, la sophistication nécessaire à l'obtention d'un certain niveau de sécurité ou de disponibilité est souvent sinon presque toujours synonyme de redondance : malheureusement l'utilisation de techniques redondantes n'est pas gratuite que ce soit en coût d'achat, coût de maintenance, poids, etc... Sachant que dans ce domaine, le mieux est souvent l'ennemi du bien, l'utilisateur, aidé bien sûr par le constructeur, devra savoir définir clairement ses objectifs prioritaires et en connaître le prix à payer dans tous les domaines de l'utilisation du matériel.

DISCUSSION

J.A.Aguer

(1) Mr Rennesson a indiqué qu'une extinction moteur sur un avion commercial ne met pas en cause la sécurité de l'avion, du faut des conditions de certification.

Je crois que cette position, bien que certifiée sur le plan de la règlementation, doit cependant être très nuancée.

La panne de moteur, au décollage en particulier, si elle entraine un arrêt de celui-ci, constitue sur les avions présents notamment les avions à grande capacité, un risque potentiel important qui nécessitera une sophistication croissante des systèmes de contrôle en particulier, qu'il est déjá nécessaire de prévoir d'ailleurs avec l'emploi des performances de secours on contingency rating.

(2) La sophistication croissante des systèmes devra néanmoins faire l'objet d'études et d'essais extrêmement poussés si l'on veut éviter que la sophistication recherchée se traduise par une baisse générale et fictive de la fiabilité basique du système – en raison des pannes inhérentes aux chaînes de contrôle ajoutées au nom d'une plus grande sophistication. Cette remarque est basée sur les modes de pannes sur an avion très récemment mis en service.

Réponse d'auteur

(1) Je suis d'accord avec M. Aguer: un arrêt en vol, même sur un multimoteurs n'est pas un évènement négligeable (il est considéré comme panne mineure en vol et panne majeure au décollage). La fréquence permettant de maintenir la double panne (panne critique à catastrophique dans les limites permises par le réglementation se situe aux environs de 2 par 10 000 heures de vol.

J'ai d'ailleurs précisé dans mon exposé que pour les moteurs civils, une panne moteur n'affecte pas dangereusement la sécurité *à condition qu'elle ne soit pas trop fréquente*, c'est-à-dire à condition que la probabilité de la double panne rest conforme aux impératifs réglementaires.

Mon propos était surtout d'indiquer que sur un monomoteur de combat, même un tel taux est inacceptable et qu'il faut pour ces matériels des systèmes de régulations beaucoup plus sophistiqués pour atteindre le niveau de sécurité requis.

(2) La remarque de M. Aguer rejoint la conclusion de mon propos qui visait surtout les utilisateurs militaires mais qui est tout aussi valable pour le matériel civil.

La seule différence est que pour ces derniers, le niveau de sécurité est défini, non par l'utilisateur mais par la réglementation. Dans ce cadre, c'est donc au constructeur, responsable de la conception vis-à-vis des Autorités Certifiantes, de définir le niveau de complexité nécessaire à la satisfaction du règlement: il est tout aussi évident que pour les matériels actuels, cette démarche ne peut se faire qu'avec l'avis et la collaboration de l'utilisateur.

A PROCEDURE FOR PREDICTING THE LIFE OF TURBINE ENGINE COMPONENTS

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SUMMARY

This paper describes a procedural method to follow for the creation of a life estimate of aircraft gas turbine engine components. The method consists of three segments - the calculation of a modulus, the determination of a critical material property, and a comparison of the modulus to the material property with a resulting judgment. Each segment is discussed in qualitative terms and related to required validation and acceptance testing.

INTRODUCTION

Reliably predicting the functional life of an advanced aircraft turbine engine component is at best a difficult task. The difficulty lies in both calculating the effect of the interaction of complex component geometry and material with the severity of the engine's environment through its diverse operational envelope and resolving the calculations into a life estimate. Despite the seemingly high complexity of the techniques employed in conducting a component life prediction, there is one underlying procedure employed. This procedure consists of three segments - the calculation of a modulus, the determination of a critical material property, a comparison of the modulus to the material property with a resulting judgment. This paper will discuss current accepted techniques used in the accomplishment of these three basic segments and relate them to required validation and acceptance testing.

BACKGROUND: THE GENERAL PROBLEM

During the past quarter century of aircraft turbine engine development, engine component life has grown from approximately 5 hours of service life to an average engine overhaul limit of 1000 hours. Life commitments on advanced engines are approaching 4000 hours on "cold" components (not in the hot gas stream) and about 2000 hours on "hot" components (in the hot gas stream).[1] These service life advancements have been achieved through the combination of new materials and advancements in design technology. However, design demands are rapidly overtaking design abilities and the increasing life trends for many components are being reversed.

The increasing design demands have exhibited themselves through an increase in both the severity of the engine component's environment and the component's operating stress level. For example, since 1955, the average tangential stress level of turbine disks at maximum engine operating speed has more than doubled. This trend is shown in Figure 1. This typical increase in disk stress level has occurred due to design criteria which have resulted in complicated geometry, high rotational speed, high differential pressure and by a high thermal gradient and operating temperature. As a result of these changing design criteria, engine disks are becoming, in general, reduced in service life through their susceptibility to failure through low cycle fatigue (LCF) as opposed to the more historical strength failure modes of creep or stress rupture.

Accurate knowledge of which failure modes are likely to be dominant for a given component is of paramount importance in the life analysis of all the engine's mechanical-structural systems. For example, one must be aware that low cycle thermal fatigue would be a major problem with dilution cooled combustors as well as thermal distortion, buckling, oxidation, and burning in order to design and estimate the life of a combustor accurately. The diversity of life considerations for an engine are shown in Figure 2. These life considerations illustrated are by no means complete, but they demonstrate the generic classes of life considerations, from inlet to exit, of a modern engine that requires investigation during the engine design cycle. In addition, in the determination of the service life of an engine component, one must be aware that many failure modes are so dependent upon the engine usage, that a new life analysis must be accomplished for each new flight vehicle application.

The problem of reduced engine component service life can be solved by utilizing the proper procedure for doing life prediction during the engine design process. This paper will discuss a procedure to follow in order to create a reliable engine design through accurately designing and assessing the life of the components.

THE PROCEDURE

To put the procedure for life prediction of engine components in perspective, consider the basic engine design cycle illustrated in Figure 3. The engine design cycle usually consists of eight phases and starts with a customer defining the desired aircraft mission(s). From this data, the propulsion requirements are derived which result in a preliminary aero-thermal design of the engine. Based upon this preliminary design configuration, the total engine, component-by-component, is structurally designed, analyzed and each component's functional life estimated. This is the portion of the engine design cycle (shown in the dotted circle) where the procedure for component life prediction characterized by this paper is utilized. Once an acceptable component and engine life has been verified through testing, the engine is ready to be

initiated into service usage. The time normally required to reach this point in the design cycle is ten years. At this point, however, requirements of the initial aircraft mission have normally been altered to the point where the engine must traverse the design cycle once again, but in far less detail.

It must be remembered in reviewing this simplified representation of the engine design cycle that there is a continuous looping of information between each phase which must stay flexible. Additionally, constant alertness must be maintained to insure that conflicts do not occur between component life goals and flight vehicle applications or aircraft missions. One must also maintain a firm understanding of the effects of analysis or data scatter in one phase on the results of other phases.

The design system must also consider a specific overhaul and maintenance schedule as a design goal in terms of integral orders of component lives. Failure to accomplish this task properly could result in an unacceptable propulsion system in terms of logistic support and life cycle cost. The need to consider life cycle cost in design can be well illustrated by comparing component replacement costs as a function of design year. As shown in Figure 4, the cost of titanium disks, based upon a 1960 price, will more than double by 1980. Also, by 1980, the cost of nickel base disks will increase by fivefold. Thus, the designer must consider performance, life, schedule and cost more equally today than ever before in creating a propulsion system.

PROCEDURE SEGMENT #1

The first segment of life prediction, the calculation of a modulus, is basically the "structural analysis" phase of the engine design cycle (Figure 3). The three most important moduli are stress, strain and energy. The calculation of these moduli require an estimate of the intended engine service usage to be made so that the analyses are conducted under the properly chosen boundary conditions. The analysis calculations are best derived through the application of finite element type computer programs exemplified by the NASA structural analysis program, NASTRAN. These types of computer analysis programs do yield precise values of stress, strain, energy, deflection, etc., provided precise values of boundary conditions are available and used for input. The need for precise boundary conditions cannot be overemphasized; for when they are in error, the analysis will be in error regardless of which computer program or analysis method is utilized.

The term "boundary conditions" includes all pressures, temperatures, forcing functions, rotational and directly applied forces on a component. Boundary conditions do not include material properties or the variation in component geometry and material properties as a result of manufacturing tolerance and processing. Usually, boundary conditions must be estimated for the first iteration of all stress analyses. However, after the engine component is tested under the proper conditions with instrumentation, the boundary conditions will be more precisely known and must be used in a refinement of the original stress analyses. Thus, the correct analysis technique is to continually refine the stress analysis as more precise definition of the boundary conditions evolve. Ideally, this process should be carried out for each component. However, in today's engine development environment of limited time and resources, only the critical components can be analyzed in this detailed, iterative fashion. The components to be considered as critical are the rotating components such as disks, blades and shafts which produce high levels of kinetic energy during engine operation. Other major components such as stationary stator vanes, combustors and cases can usually evolve with somewhat less precise analysis if proven conservative during qualification testing.

The analysis techniques utilized should be compatible with the state of stress incurred by the component. For locations of high stress (areas where material plastic or creep flow will occur), a detailed inelastic solution technique must be employed. For component areas that are cyclically strained, a solution must be obtained for each set of cyclic boundary conditions experienced. Additionally, the analysis technique utilized must be applicable to the class of structure being analyzed. One needs to use shell finite element programs on shell-like structures such as combustors and cases and solid, three-dimensional element programs on critical solid structures like disks and blades. Care must be exercised in each analysis that a cost effective solution is achieved in the time period where it will impact the design process most meaningfully. It does little good to try to optimize the design of a blade dovetail after manufacturing release has been given for the disk.

The most difficult facet of an analysis is the determination of a design "duty cycle" from a matrix of possible flight missions. One accepted method of accomplishing this task is to simply survey existing, similar engine usage. Data recorded for non-dwell effected components during the survey would include rpm excursion (percent of maximum) and frequency of rpm excursion occurrence for the class of engine activities estimated for the new design. Typical military engine activities to be considered in the survey should include:

- Take-Off
- Landing
- Touch-and-Go
- Air Maneuvers:
    - Air-to-Air Combat
    - Air-to-Ground Combat
    - Bomb Delivery

From the survey data, one would pick the most severe magnitude and frequency of throttle movement for each activity and use it as a design subcycle. Each subcycle would be normalized on frequency of occurrence to one flight and combined in proper order to produce a design duty cycle. Such a procedure is illustrated in Figure 5 and Figure 6.

In Figure 5, three subcycle flight activities are shown on the left of the figure for two different configurations of the same engine model. The data represents a 100 flight survey in terms of percent N_1

throttle excursion. Based upon this data, a set of design subcycles were constructed for the non-dwell effected components. On the right of Figure 5, the chosen design subcycles are shown. In Figure 6, these subcycles are shown combined together to form the actual design duty cycle on a one flight basis. This type of duty cycle creation is acceptable for non-dwell effected components where induced stresses from the subcycle activity are governed primarily by centrifugally created forces.

For components which are dwell effected and/or exposed to high (> 40% absolute material melting temperature) operating temperatures, the rpm effects are not as dominant and the stress variations resulting from engine rpm changes are difficult to predict without knowledge of the effects of the incurred thermal gradients. For these components that are temperature and temperature gradient affected, one must also record the dwell time and temperature associated with the subcycle activity. One must have this knowledge so that the created design duty cycle for the dwell affected components is realistic in terms of creep time and stress relaxation allowances.

During the analysis of each component, there is a very important but frequently overlooked life analysis consideration. This consideration includes the secondary flow (cooling air flow) and resulting heat transfer analysis as it is affected by the engine's structural response to each duty cycle throttle motion. This secondary flow analysis must be conducted because even subtle hardware deflection can cause localized flow and temperature changes to occur which can have a substantial influence on the life of many critical components. The secondary flow analysis is very difficult to accomplish due to the inadequacies of currently available fluid flow models to predict 3-dimensional fluid heat transfer behavior. As a result of the limitations of analysis, detailed testing should be accomplished to insure that secondary flow effects are thoroughly characterized for each component. Failure to do the proper detailed testing and analysis of the secondary flow effects will result in a component design that is high risk in terms of achievement of design life.

PROCEDURE SEGMENT #2

The second segment of life prediction is the determination of a critical material property. This occurs during the "material characteristics" portion of Figure 3, the engine design cycle, and must consider the environment produced in the engine as it responds to its duty cycle throttle settings.

For static structures, the property may be yielding, ultimate strength or fracture toughness at a particular operating temperature. For rotating components, it may be endurance stress (fatigue), ultimate stress, .02% yield stress, fracture toughness or any combination of these properties.

Regardless of which material properties are chosen, there will be data scatter associated with them that must be taken into consideration. The consideration of the scatter is in reality a determination of the acceptable failure rate of the material in a chosen mode. A currently accepted practice is to assume a normal material property distribution and to use a material property that has been reduced from the average of empirical data by three standard *deviations* (-3σ). This translates into 99.87% of the population having a value in excess of the chosen property level.

It is very important that effects of environment and manufacturing processes be considered in the choice of the material properties as they usually have a pronounced effect. These effects are especially severe on the fatigue characteristics of engine utilized materials. In a like manner, care must be exercised during the determination of the critical material property that consideration is given to heat-to-heat variation and vendor-to-vendor variation. Additionally, one must insure that he has an understanding of the effect of geometry upon the properties. For example, for thin wall cast airfoils, one needs to test thin wall cast specimens for the properties rather than standard smooth and notched bar type specimens.

The key to life prediction accuracy is the proper interpretation of the empirical material data; the better the designer understands the materials' load carrying characteristics, the more accurate his life prediction will become. The interpretation of the data is fundamental in the final procedural segment of life prediction, procedure segment 3.

PROCEDURE SEGMENT #3

The third segment of life prediction is the comparison between the results of segment number one and segment number two. This comparison is the "life prediction" phase in Figure 3 and is best accomplished through empirically established and component correlated "life curves". Three types of curves are qualitatively represented in Figures 7, 8 and 9. In Figure 7, a qualitative LCF S-N curve is shown for a constant mean stress value. As can be seen, an error in stress (strain) range determination or scatter in the material can separately or jointly cause a wide range in predicted life. In Figure 8, a qualitative high cycle fatigue (HCF) Goodman Diagram is shown. Again, as in the LCF analysis, errors in the stress calculation can produce a wide range in predicted life. In the determination of rupture life, shown qualitatively in Figure 9, the effect of stress calculation errors and environment can again separately or jointly cause a wide range in predicted life. These curves repeatedly illustrate the importance of both an accurate stress analysis based upon accurate boundary conditions and an accurate characterization of material.

In a life prediction, however, the actual determination of life is derived from the application of empirically verified failure theories which relate a multi-axial state of stress to a uniaxial state of stress for which failure test data has been determined. For brittle materials (elongation < 3% in 2 inches) under static loads, the most common failure theory used is Maximum Normal Stress Theory (Rankine's theory)[2]. For ductile materials, (elongation > 3% in 2 inches) under static loads, the most common failure theory used is the Distortion Energy Theory (Huber - von Mises - Hencky Theory)[3]. The 3% elongation criteria is only an approximation and will vary from material-to-material.

There are, unfortunately, very few components within the engine which operate under either static or quasi-static loading conditions. The majority of loads are repeating, fluctuating and rapidly applied. As a result, the majority of engine life prediction centers around the calculation of damage accumulation from fatigue. Two simple, well demonstrated damage accumulation models to use are the Palmgren-Miner Linear Damage Theory[4] and the Manson Double Linear Damage Rule[5]. These two methods operate upon the standard S-N constant amplitude curves which are usually readily available or relatively inexpensive to generate. Newer methods like Strainrange Partitioning[6] show promise in this technical area and are currently being evaluated on a broad basis throughout the world. The Strainrange Partitioning method involves the determination of four basic life relationships for the four possible cyclic combinations of material plastic flow and creep flow and their use in conjunction with an interaction damage rule to predict life limits. Detailed information on this method is available in Reference 6.

Throughout the turbine engine industry, there are many other methods of assessing damage and graphically presenting a material's behavior. However, to effectively use them requires in-depth knowledge of a particular design system and in the past, they have shown no particular increase in life prediction accuracy.

LIFE PREDICTION VALIDATION THROUGH ACCEPTANCE TESTING

The accuracy of the life predicted for any engine component is very difficult to demonstrate without a priori knowledge of full fleet experience. However, a manufacturer must be able to demonstrate a required "durability level" by successfully completing a number of severity tests upon his component and engine design. Experience has shown that undue hardship testing does not demonstrate or insure durability and life for each engine component. This is due to the fact that the severe testing conditions do not apply severe boundary conditions to each engine component in a direct proportion to field usage. What needs to be done in order to properly establish and utilize severity testing is to establish the local response of critical areas of each component to the precise boundary conditions produced. This is the only way to meaningfully interpret the data from severity testing in terms of validating a life prediction number or method. Unfortunately, the type of highly instrumented testing required is both very costly and time-consuming. Therefore, whenever possible, bench or rig testing should be conducted in lieu of full engine testing to characterize the inherent structural life of the critical components. The full engine test should be the vehicle where boundary conditions are measured for refinement of stress and life analyses, not the demonstration of actual life.

CONCLUSION

An engine manufacturer cannot non-destructively inspect quality, durability and long life into an engine component; they have to be created through careful design and manufacture. The level of success in achieving these desired qualities depend upon the designer's ability to iteratively refine stress analyses, empirically characterize his material under all applicable conditions, and, with verified repeatable techniques, compare the analyses to the data and obtain an acceptable life determination. The key to each is the identification of the intended duty cycle and, most importantly, the identification of accurate boundary conditions as a function of the duty cycle.

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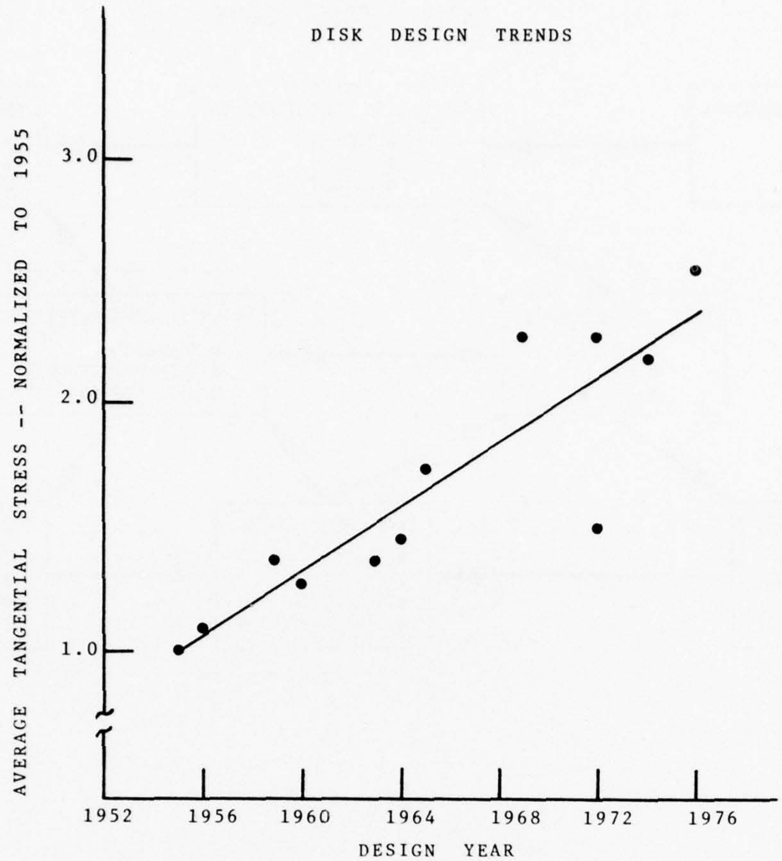


FIGURE #1

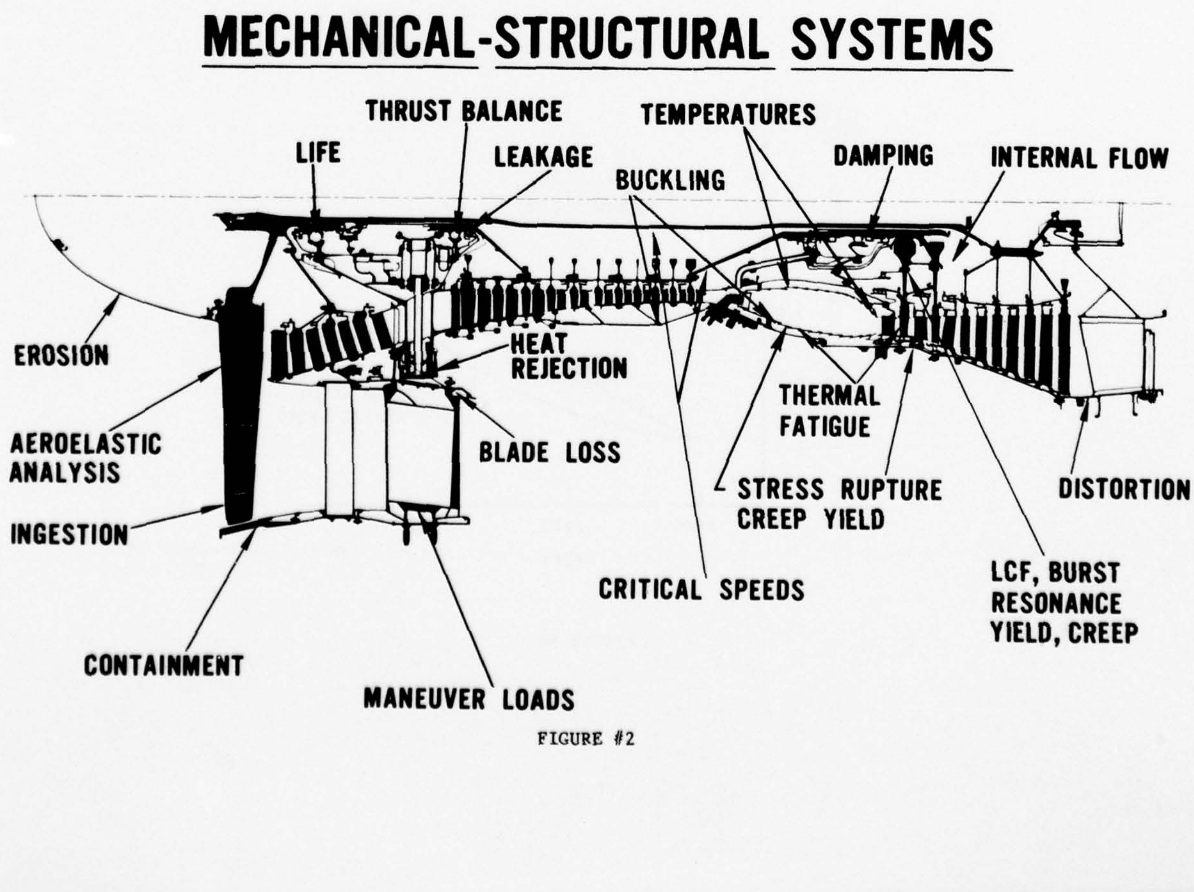


FIGURE #2

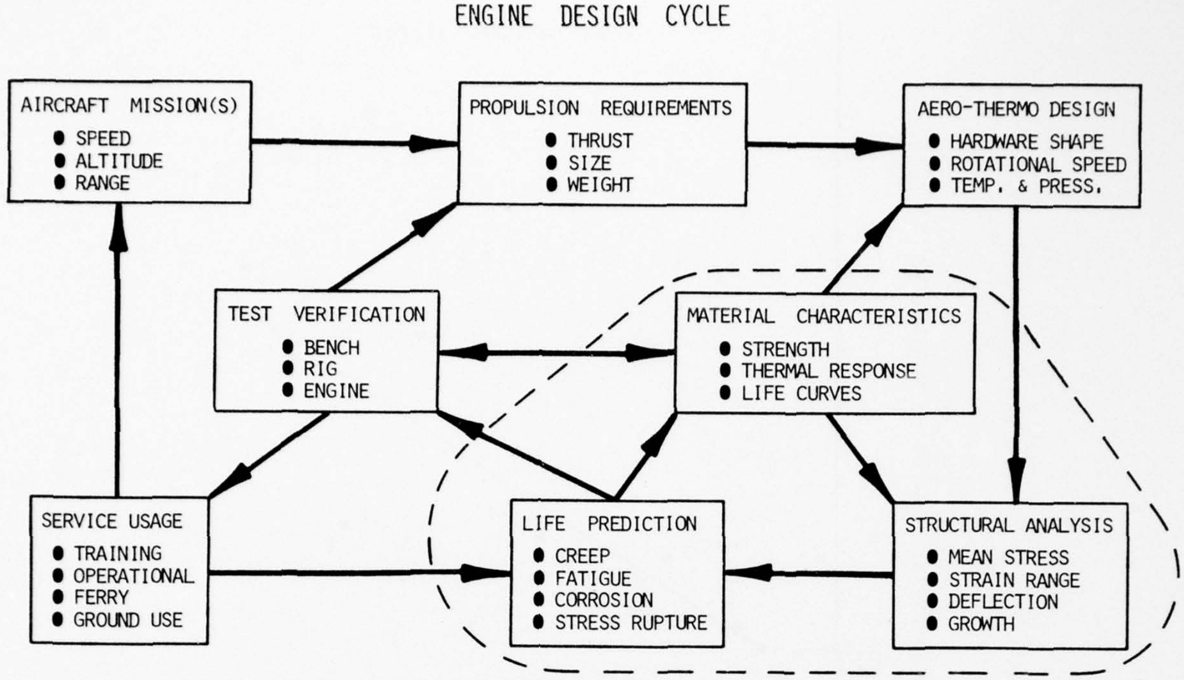


FIGURE #3

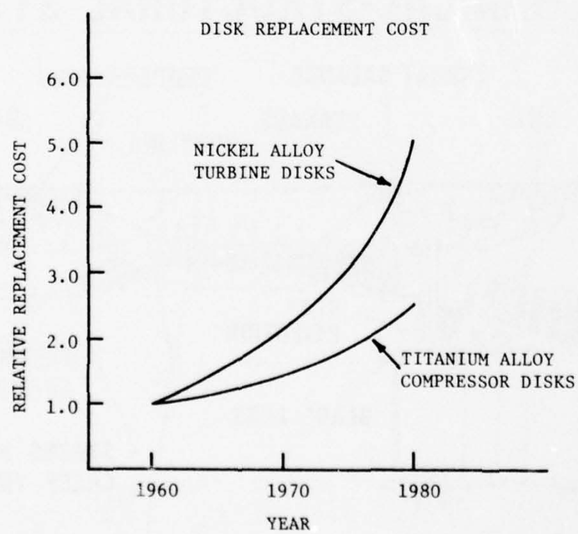


FIGURE #4

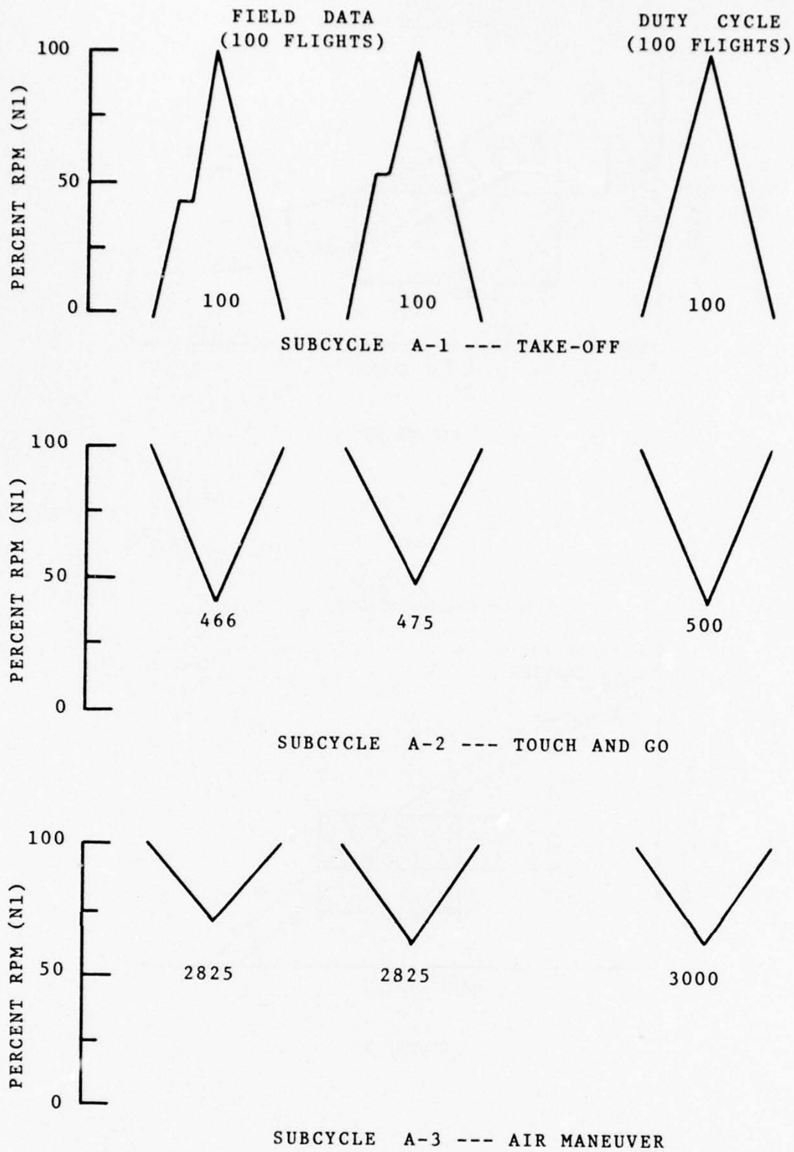
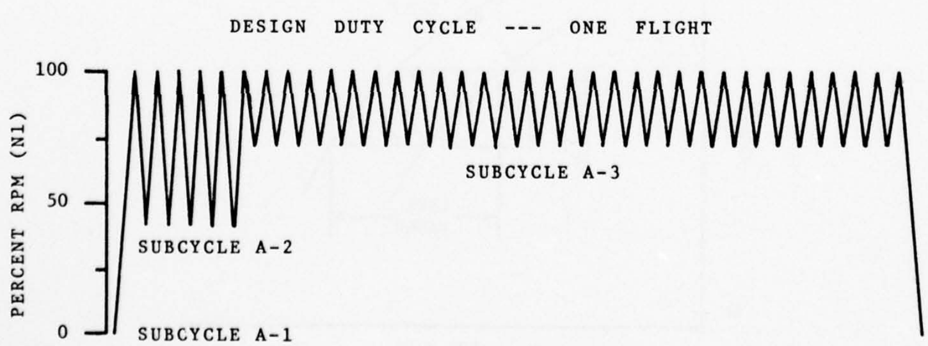


FIGURE #5



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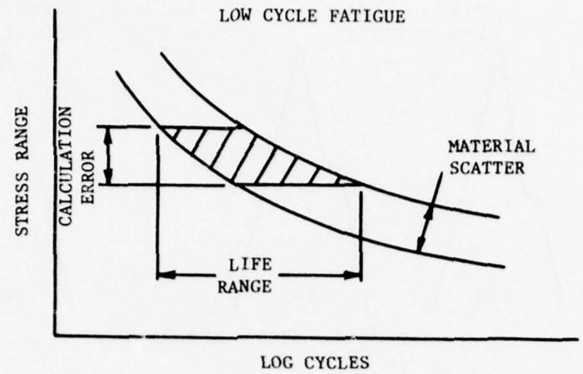

FIGURE #7

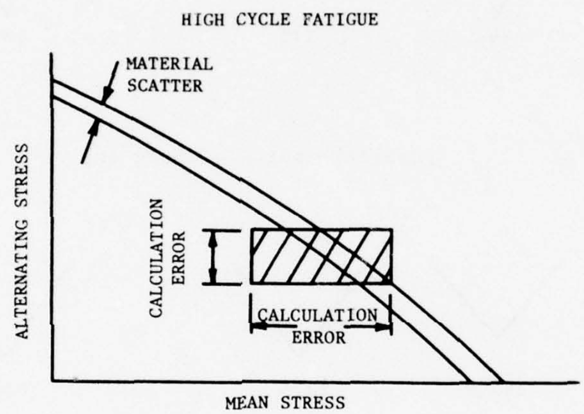

FIGURE #8

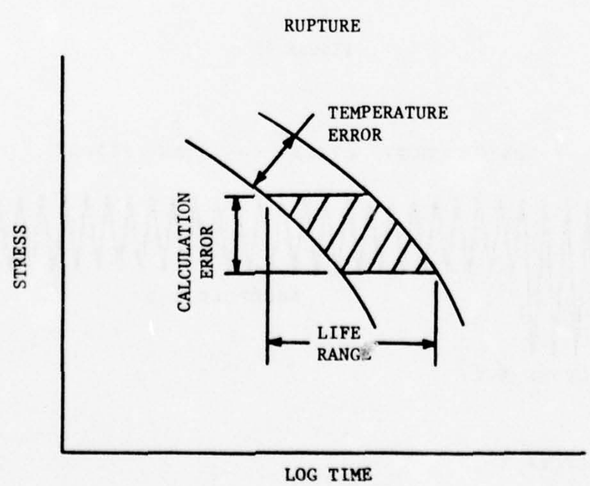

FIGURE #9

DISCUSSION

J.C.Ripoll

How would you consider carrying out tests in accordance with the duty cycle defined on Figure 6?

Would you use rigs (components) or engines?

Author's Reply

The cycle I have shown in my figure represents a cycle for the non-dwell affected components. These are components which are primarily dominated by centrifugal effects and not temperature effects. When one considers dwell affected components (which operate above 40% of the absolute material melting temperature) one must modify that cycle to include dwell time, time at temperature such that proper creep and stress relaxation is allowed to occur during the testing. The figure I was showing in my slide would be used for testing cold components, fan or compressor components. You would perform these cycle tests on a rig basis first and if this proves the analysis then you could run the same type on a full engine.

EVOLUTION ET CONTROLE DES DIFFERENTS PROCESSUS DE DEGRADATION DE PERFORMANCE SUR LES PROPULSEURS MODERNES

par
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RESUME

Nous rappelerons d'abord quelques généralités en ce qui concerne l'évolution de la politique d'entretien des réacteurs, et les principes d'application fondamentaux des méthodes de monitoring, puis nous examinerons successivement les évolutions des processus de dégradation de performances d'ordre thermodynamique et d'ordre mécanique. Enfin, nous mentionnerons un exemple du type de contrôle qui permet d'évaluer l'efficacité de ces méthodes, tant du point de vue technique que du point de vue économique.

1. GENERALITES

1.A. Evolution de la politique d'entretien des réacteurs

Etapes fondamentales

On peut distinguer plusieurs étapes fondamentales dans l'évolution de la politique d'entretien des réacteurs :

- de 1959 à 1963, les méthodes d'entretien consistaient essentiellement à effectuer des révisions générales à potentiel fixe.

- de 1963 à 1966, furent pratiquées des révisions spécifiques des parties froides et des parties chaudes des réacteurs, avec introduction de visites intermédiaires.

- Ce n'est qu'en 1966 que furent introduits les programmes de fiabilité de propulsion qui donnaient une place importante aux méthodes de monitoring.

- En 1969, l'entretien se spécialise par sections (entretien modulaire), et enfin, en 1972, les potentiels fixes (de section) font place à la notion de seuils de mise en réparation (entretien selon état).

Toutes ces méthodes contribuèrent à l'optimisation de la fiabilité "mécanique".

- En 1975, la notion de méthodes complémentaires d'entretien, nécessaires pour rétablir les performances, s'impose, notamment à la suite de la crise économique de 1973, qui voit le prix du carburant augmenter considérablement, ce qui donne une importance accrue au rétablissement des consommations spécifiques, en particulier.

Objectifs

L'objectif essentiel de la politique actuelle d'entretien des réacteurs consiste à surveiller la fiabilité de propulsion et à améliorer la fiabilité intrinsèque de conception par une évolution des standards. A ceci s'ajoute la nécessité nouvellement ressentie de prévenir la détérioration des performances, par la mise en oeuvre de méthodes intégrées de surveillance continue sur avion. L'ensemble des objectifs ainsi résumés, contribue à optimiser les coûts opérationnels et de révision. (Réf 1)

Structures

Les structures de cette politique d'entretien consistent essentiellement à :

- maintenir les critères généraux de fiabilité précédemment en vigueur,

- choisir les méthodes de surveillance sur avion en fonction des problèmes et de la technologie des réacteurs,

- développer l'efficacité des méthodes de surveillance sur avion et de vérification au sol,

- optimiser la période d'intervention en atelier pour pratiquer, en particulier, le reconditionnement du passage des gaz.

Ces structures supposent un recours intensif aux méthodes de monitoring, et l'on peut affirmer en résumé, que le choix, le développement et la mise en oeuvre des programmes de monitoring sur avion, présentent désormais une importance fondamentale dans les techniques les plus modernes d'entretien des réacteurs. (Réf 2, 3, 4)

1.B. Principes fondamentaux d'application des méthodes de monitoring

. Deux phases essentielles marquent le développement de ces méthodes.

- Enregistrement des données :

  L'étude de l'historique des réacteurs ayant subi une panne de même nature permet de dégager dans les différentes évolutions des paramètres surveillés, ceux dont l'allure, antérieurement à la panne, présentent des caractères communs. Un cahier de signatures de pannes est alors établi.

- Reconnaissance des pannes et lancement des actions de corrections préventives :

  La comparaison avec les références établies ci-dessus des comportements ultérieurs des réacteurs, permet le diagnostic du défaut et le lancement d'actions de corrections appropriées.

. Le signal d'alerte doit être progressif.

Seuls sont susceptibles de surveillance par une méthode de monitoring, les incidents pour lesquels un paramètre bien déterminé dévie de façon caractéristique et progressive.

La figure 1 résume un processus typique. Lorsque le signal présente un écart caractéristique par rapport à la zone de dispersion normale du phénomène, l'anomalie est décelée. Elle est ensuite confirmée et la mesure de correction programmée. Cependant, le lancement de cette action corrective s'applique avec un certain retard, dû à la transmission et à l'analyse des données. Si l'incident grave, en l'absence de mesures correctives, se produit au temps B, l'application de la mesure corrective aurait dû se faire en un temps A inférieur.

La figure 2 donne l'exemple du monitoring d'un réacteur JT9D7 sur B.747 où l'on voit une déviation caractéristique des paramètres température de gaz et débits carburant. L'anomalie décelée a été confirmée par observation boroscopique, elle résulte d'une diminution du jeu en bout d'ailettes de la turbine HP, provoquée par un départ de métal dans la veine des gaz, à la suite d'une importante sulfidation en extrémité de ses ailettes.

La figure 3 résume, pour les moteurs OLYMPUS du Concorde, différentes évolutions des taux de pollution de l'huile en particules de fer. Cette pollution se produit par suite d'un phénomène d'usure qui prend place sur certaines cannelures de l'arbre de commande du boîtier d'entraînement des accessoires. Celles-ci sont elles-mêmes lubrifiées par l'huile de graissage normal des roulements principaux du réacteur. Le phénomène s'accélère rapidement et peut conduire à la disparition des cannelures. Les incidents graves (arrêt du fonctionnement réacteur) sont représentés avec un fléchage noir.

On voit sur ce dernier exemple que la vitesse à laquelle croît l'usure, impose de faire des prélèvements d'huile toutes les 15 heures environ, et d'avoir le résultat presque immédiatement si l'on veut maîtriser le phénomène, dans sa durée de développement la plus fréquente.

Ces exemples illustrent le principe général suivant : la durée minimum de développement du défaut avant incident grave, détermine l'organisation d'application pratique de la méthode des points de vue

- fréquence de collecte
- délai maximum acceptable de transmission, traitement et analyse des données recueillies.

2. EVOLUTION DES METHODES DE SURVEILLANCE DES PROCESSUS DE DEGRADATION DES PERFORMANCES REACTEUR

2.A. Dégradations d'ordre thermodynamique

Trois grandes étapes peuvent être distinguées :

- de 1959 à 1963, on se contentait de vérifier le niveau des performances, au banc d'essai sol, lors des révisions générales.

- de 1963 à 1975, le suivi journalier des écarts de performances individuelles des réacteurs par rapport à un réacteur de référence a été généralisé et s'est vu complété par un suivi périodique des performances moyennes statistiques de la flotte. Toutefois l'apparition des appareils gros porteurs (B.747, DC-10) ne s'est pas généralement accompagnée du développement attendu des méthodes AIDS.

- depuis 1975, une instrumentation appropriée a été introduite, ainsi que des méthodes d'analyse particulière pour l'évaluation des performances modulaires. En effet, la recherche d'une meilleure efficacité semble avoir conduit davantage à la localisation des chutes de rendement, plus qu'au suivi continu de leur évolution, par système d'enregistrement embarqué. (Réf 5, 6 et 7)

Les figures suivantes illustrent ces différentes étapes.

. 1ère étape : Relevé des performances au banc

La figure 4 est relative au cas du suivi du niveau de puissance au point fixe, d'un turbopropulseur DART entre sa sortie de révision générale et la fin de son potentiel. Deux restaurations intermédiaires des performances ont pris place, l'une sur avion avec nettoyage compresseur, l'autre

par révision des parties chaudes en atelier et réaffichage de la puissance maximum, au prix d'une augmentation de la température de fonctionnement.

. <u>2ème étape</u> : <u>Suivi des performances en croisière</u>

Pour la deuxième étape, aux évaluations précises effectuées au banc d'essai, s'ajoutent les indications relatives du suivi des performances en croisière. Dans cette méthode, à ATLAS* par exemple, les données sont recueillies par le mécanicien navigant, transmises par télégramme à un ordinateur central depuis la première escale touchée, stockées, puis traitées automatiquement chaque nuit par un programme de réduction en standard et de lissage mathématique. Les listings produits sont analysés dès le lendemain matin par l'ingénieur de maintenance. Le délai entre collecte des données, analyse et lancement des mesures correctives, est actuellement d'environ une dizaine d'heures (au maximum, de 24 heures). Il permet d'atteindre couramment une efficacité relativement élevée, dans la reconnaissance des défauts se développant à moyen terme. Un essai limité AIDS (Air Integrated Data System) n'a pas encore permis d'améliorer cette situation, <u>dans ce domaine</u> du moins, au prix cependant d'un investissement assez important.

Les figures 5, 6 et 7 reproduisent les applications pratiques du suivi des performances en croisière, élaborées par Pratt & Whitney, General Electric et Rolls Royce pour les moteurs JT9D sur B.747, CF6-50 sur DC-10 et A-300, OLYMPUS sur CONCORDE.

Les données recueillies en croisière sont également analysées statistiquement, en les groupant par type d'installation. La figure 8 représente l'évolution, au cours des années 1974, 1975 et 1976, des performances moyennes flotte des réacteurs JT3D sur B.707 à Air France. Chaque point représente la moyenne statistique d'environ 2500 relevés.

La figure 9, enfin, représente des exemples de suivis à très court terme, effectués au cockpit par les mécaniciens navigants dans le cas du CF6-50. L'incident est relatif à la rupture des boulons d'attache du carter intérieur de chambre de combustion du réacteur CF6-50. Il se traduit par une augmentation brusque de la température des gaz. Il est demandé dans ce cas, de réduire la puissance pour éviter des dégâts secondaires très importants (ils peuvent se monter à plusieurs centaines de milliers de dollars) consécutifs à la surcharge du roulement de butée du mobile HP, et à sa défaillance qui entraîne, en particulier, l'interférence entre rotor et stators de compresseur HP et leur destruction.

. <u>3ème étape</u> : <u>Performances modulaires</u>

L'évaluation des performances modulaires devient impérative pour contrôler leur dégradation dans le temps et organiser un programme de reconditionnement approprié.

Pour atteindre ce but, il a fallu introduire aux plans intermodulaires une instrumentation complémentaire et mettre en jeu des méthodes d'interprétation appropriées (vectorielles chez PWA, analytiques chez G.E.). (Réf 5)

Sur avion, il semble également souhaitable de pouvoir assurer la permanence de la surveillance précédente. Certaines solutions sont déjà prévues qui font appel à la notion de transfert d'information (sous forme électrique) à une zone déterminée de l'avion, où peut se trouver un enregistreur magnétique (PIC = Plug-in-Console ou AIDS).

En effet, à ce stade, le <u>volume des informations</u> à recueillir rend obligatoire leur collecte automatique. Il est à noter que ce type de suivi n'est pas envisagé en continu, mais à partir d'un échantillonage judicieux dans le temps, l'information étant alors stockée dans une banque de donnée centralisée, pour pouvoir être analysée ensuite en différé.

Il faut noter enfin que la possibilité, essentielle pour l'utilisateur, de pouvoir exécuter cette surveillance, nécessite au préalable la collaboration étroite du constructeur moteur et de l'avionneur, pour définir les buts poursuivis et les méthodes d'analyse employées parce que cela suppose des moyens d'investigations tout à fait particuliers.

* Nota : ATLAS est le consortium formé au niveau de l'entretien par la réunion des compagnies aériennes AIR FRANCE, IBERIA, LUFTHANSA, ALITALIA, SABENA.

En résumé, l'organisation du suivi de la dégradation des performances réacteurs apparaît comme devant désormais être envisagée dans un ensemble intégré à 3 niveaux, du type que nous avons appelé EMIS (Engine Monitoring Integrated Systems).

niveau	objectif essentiel	caractéristique de la déviation	surveillance assurée par .../...	.../... au niveau	exemples
1	améliorer la fiabilité de propulsion (réduction des arrêts en vol)	tendances à court terme (heures)	mécanicien navigant.	cockpit	suivi des élévations brusques EGT, à poussée constante
2	améliorer la régularité d'exploitation (éviter les changements réacteurs en escale)	tendances à moyen terme (jours-semaines)	analyste en maintenance	base principale de maintenance	Programmes "ADEPT", "PWA combined" .....
3	améliorer le coût à l'heure de vol (organiser les remises en état, maintenir la Cs)	tendances à long terme (mois-années)	ingénieur de révision	agence centrale de révision	Analyse vectorielle, PICS/AIDS et DATA BANK .....

2.B. Dégradations d'ordre mécanique

Les méthodes de surveillance qui permettent de suivre les dégradations d'ordre mécanique les plus courantes diffèrent, selon qu'elles s'adressent aux ensembles mobiles ou aux ensembles statiques.

. Ensembles mobiles

En ce qui concerne les ensembles mobiles plusieurs étapes peuvent être distinguées en ce qui concerne l'évolution de leur état vibratoire.

- de 1959 à 1963, la perception physiologique des vibrations au cockpit permet généralement de surveiller de façon continue l'état vibratoire (Caravelle).
- A partir de 1963, un système d'évaluation quantitative fut installé sur avion Boeing de la première génération (B.707, B.727, DC 8, DC 9). Il était fait appel à des capteurs dits "de vitesse".
- A partir de 1965, les accéléromètres, plus fiables, remplacent les capteurs de vitesse (B.747).
- En 1970, l'introduction de filtres "passe-bas" et "passe-haut" permet d'isoler le balourd sur les mobiles haute et basse pression (DC 10 et A300B).
- 1976 marque probablement une nouvelle étape grâce au recours aux filtres à bande passante "suiveuse", qui permettent d'atteindre une évaluation de l'importance du balourd et de son calage angulaire, indépendamment des caractéristiques d'amortissement des supports.

  La figure 10 donne pour le CF6-50, les allures caractéristiques des réponses d'un accéléromètre au balourd causé par des ailettes de compresseur endommagées, suite à ingestion d'un corps étranger.

  La figure 11 reproduit le spectre en fréquences d'un réacteur CF6-50 présentant un balourd caractéristique en vitesse de croisière.

. Ensembles statiques

Différentes étapes ont marqué l'évolution des méthodes de surveillance des pièces réacteur lubrifiées par huile.

- de 1959 à 1965 : seuls étaient surveillés les filtres sur les lignes de retour au réservoir, d'huile des paliers.
- en 1965, furent introduits les bouchons magnétiques,
- dès 1970, les méthodes de spectrographie de l'huile marquent une étape fondamentale dans la reconnaissance avancée des phénomènes d'usure anormaux.

Elle a été complétée récemment par la méthode ESP (Engine Soap Prediction).

La prochaine génération de réacteurs verra peut-être l'installation de circuits de détections sur le réacteur proprement dit (système ELCA de PWA, système FERROGRAPH de US Navy).

La figure 12 illustre un exemple caractéristique de suivi par spectrographie d'émission du phénomène d'usure de cannelures d'entraînement de l'arbre de commande de boîtier accessoire d'un CF6-50. La méthode de suivi étant indirecte, elle nécessite d'être complétée par une méthode de recoupement, soit conductibilité électrique de l'huile (test EPPI), soit gammagraphie.

NOTAS :

1) Influence de la consommation d'huile

A usure constante, le niveau de pollution dépend de la consommation d'huile. Lorsque celle-ci est constante, une interprétation qualitative directe est possible. Sinon il est nécessaire de tenir compte de ces variations, ce qui peut être fait par ordinateur à partir de l'enregistrement du nombre de pleins entre prélèvements, et de l'importance quantitative de ceux-ci.

La figure 13 illustre cette nécessité.

2) Influence de la grosseur des particules

L'étude de la répartition statistique, en taille, des particules de métal engendrées par le phénomène d'usure montre que celle-ci varie de façon caractéristique lorsque le défaut s'accélère et que l'incident de fonctionnement est imminent. C'est ce qui est résumé par la figure 14.

La figure 15 illustre un cas typique de JT8D qui a pu être maintenu sur avion malgré un taux de pollution relativement élevé.

3. CONTROLE DE L'EFFICACITE DES METHODES DE SURVEILLANCE

3.A. Efficacité technique

Le succès relatif des méthodes de monitoring est couramment évalué à Air France par exemple, par le nombre de descentes programmées, classée selon le premier signe d'alerte qui les a provoqué. Le niveau d'efficacité peut alors être défini par le rapport nombre total de descentes effectuées par l'utilisation des méthodes de monitoring (premier symptôme), au nombre total des descentes.

Toutefois, étant donné l'intégration des différentes méthodes de monitoring dans les programmes modernes d'entretien, et en particulier, le fait qu'elles permettent d'entreprendre des actions correctives en dehors des descentes réacteur, cette évaluation est très restrictive et ne peut prétendre, à elle seule, être totalement représentative.

La figure 16 fournit cependant, à titre indicatif, l'évaluation de l'efficacité relative des différentes méthodes de monitoring dans le cas du réacteur CF6-50 pour l'ensemble des compagnies du groupe ATLAS.

3.B. Contrôle de l'efficacité économique

. Il convient de rappeler tout d'abord deux principes fondamentaux que l'on ne doit pas perdre de vue dans la recherche de l'efficacité économique d'un ensemble de méthodes de monitoring.

- Le total des coûts de mise en oeuvre de ces méthodes ne doit pas être supérieur aux économies de dépenses correspondantes sur les coûts d'entretien à l'heure de vol.

- L'évaluation doit être faite globalement pour les méthodes de monitoring pratiquées conjointement.
  . Paramètres
  . SOAP
  . Boroscope
  . Radioisotopes

car ces méthodes sont complémentaires les unes des autres. Le recoupement est indispensable dans le cas des méthodes indirectes. Très souvent, d'autre part, les réacteurs descendus présentent plus d'un seul symptôme.

. L'expérience montre, d'autre part que, l'optimisation d'un programme intégré comprenant l'ensemble de ces méthodes, dépend largement :

- du type de réacteur sur lequel elles s'appliquent,
- de l'organisation générale du programme d'entretien,
- des contraintes particulières d'exploitation, et de la structure du réseau des compagnies utilisatrices.

La figure 18 donne l'allure générale de ces 2 paramètres en fonction du degré de complexité du système de surveillance.

La figure 19, enfin, montre les résultats qui ont pu être estimés dans 2 cas de réacteurs modernes à large taux de dilution.

Comme on peut le voir, la réponse dépend largement de l'organisation du programme d'entretien adopté.

IV CONCLUSION

En résumé, on peut affirmer de ce qui précède qu'il n'y a pas de solution unique au problème général de surveillance des processus de dégradation de performances réacteur, mais une optimisation qui, dans chaque cas, peut varier assez largement d'une application à une autre.

Cependant, les méthodes habituelles de surveillance se développent dans le sens d'une localisation des défauts décelés au niveau du module ou de la pièce, en respectant l'objectif économique essentiel d'un retour sur investissement satisfaisant.

Enfin, ce développement suppose désormais, essentiellement, une coordination étroite et préalable, entre constructeur et utilisateur des futurs réacteurs.

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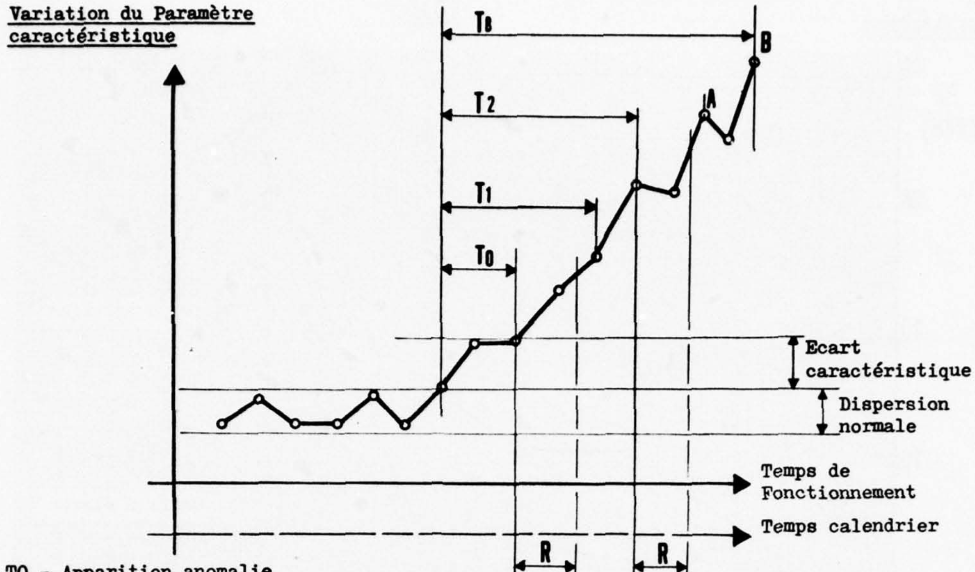
ALLURE GENERALE DES PHENOMENES A DETERIORATION PROGRESSIVE **FIG.1**



TO - Apparition anomalie
T1 - Confirmation anomalie - Programmation mesure de correction
T2 - Lancement action corrective A - Application action corrective
R - Retard dû à la transmission et à l'analyse des données
TB - Incident grave en l'absence de mesure corrective

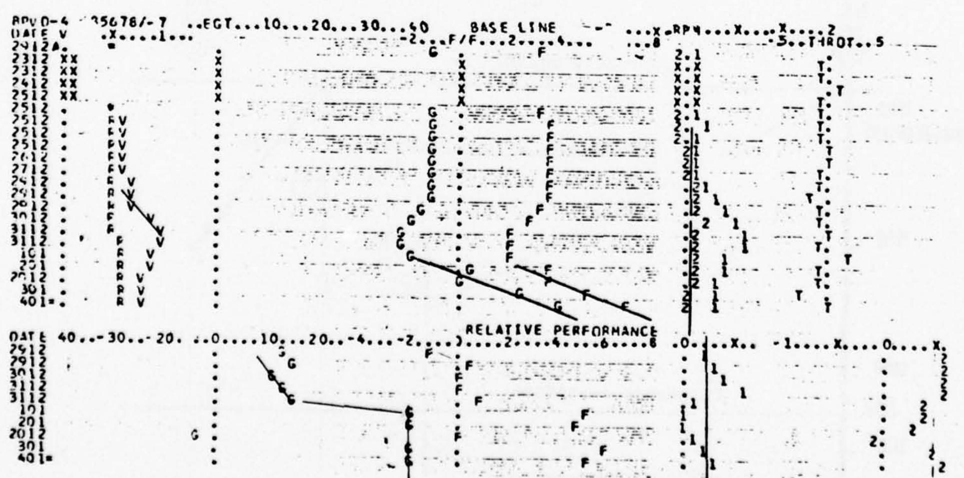
Succès MONITORING Si TA < TB

FIG.2

B.747 _ JT9D7

BPVD.4 _ 685675 9.01.75

PROGRAMME DE SUIVI PERFORMANCES EN CROISIERE

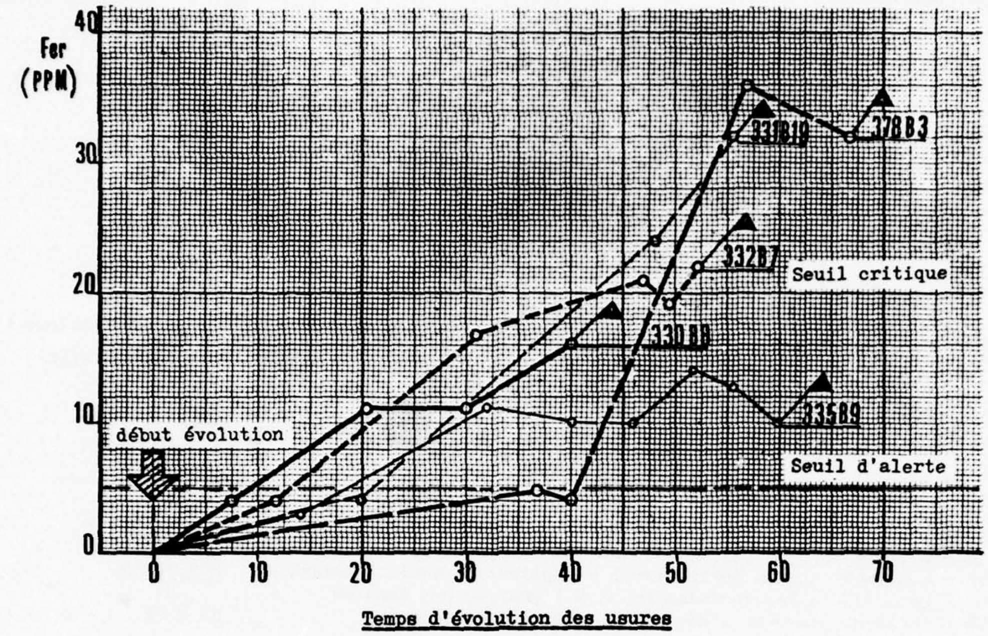


CONCORDE
RR.OLYMPUS_593

FIG.3

S.O.A.P

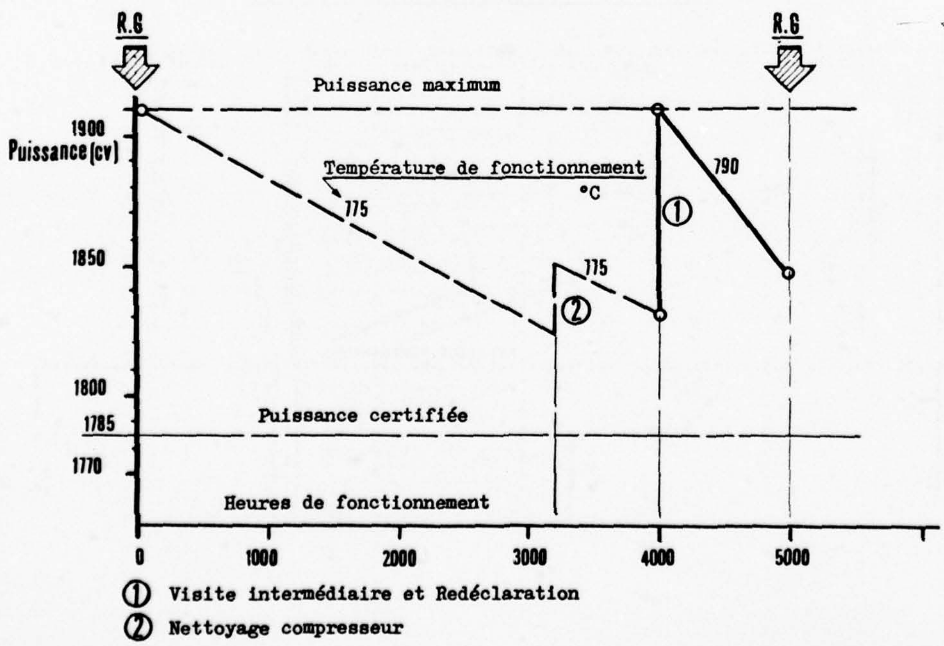
Taux de pollution



TP_DART.532_7
n° 14253

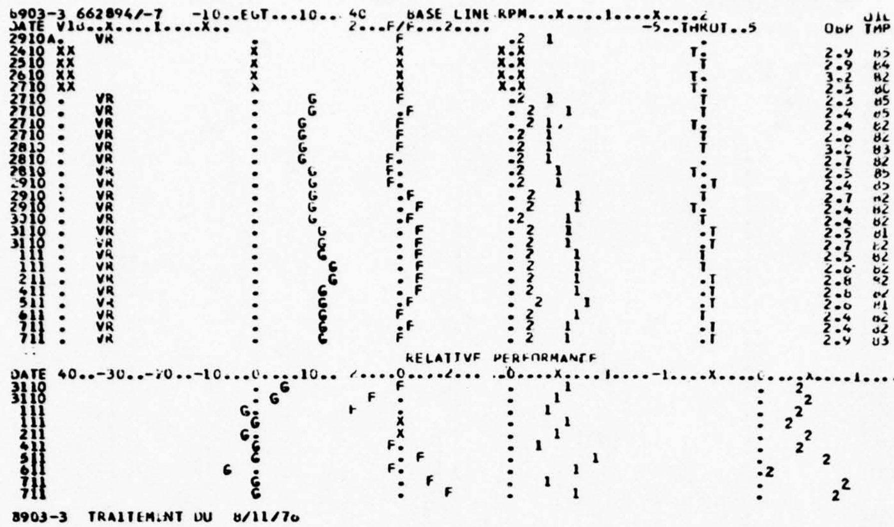
FIG.4

VERIFICATION DU NIVEAU DES PERFORMANCES AU BANC D'ESSAIS SOL LORS DES REVISIONS GENERALES



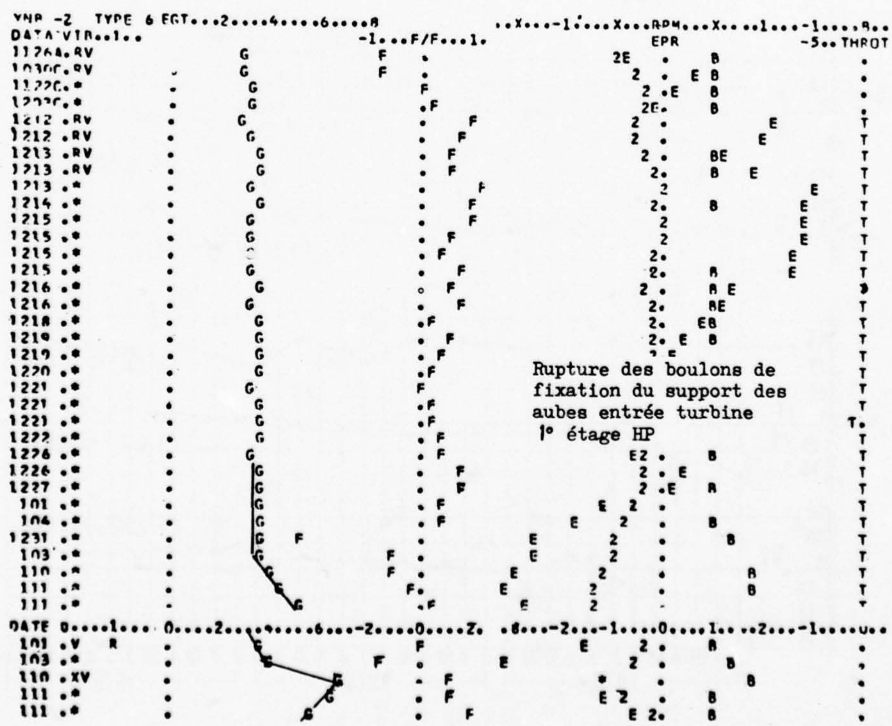
B 747_JT9D7 **FIG.5**

n°662894



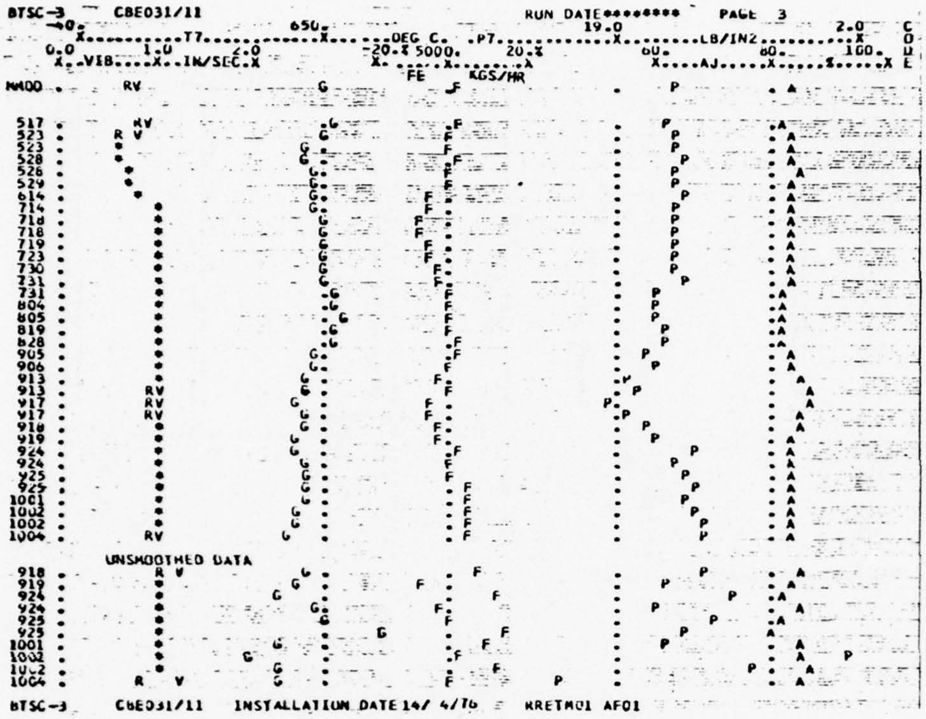
DC.10 CF6.50 **FIG.6**

n°455418



RR.OLYMPUS_593 FIG_7

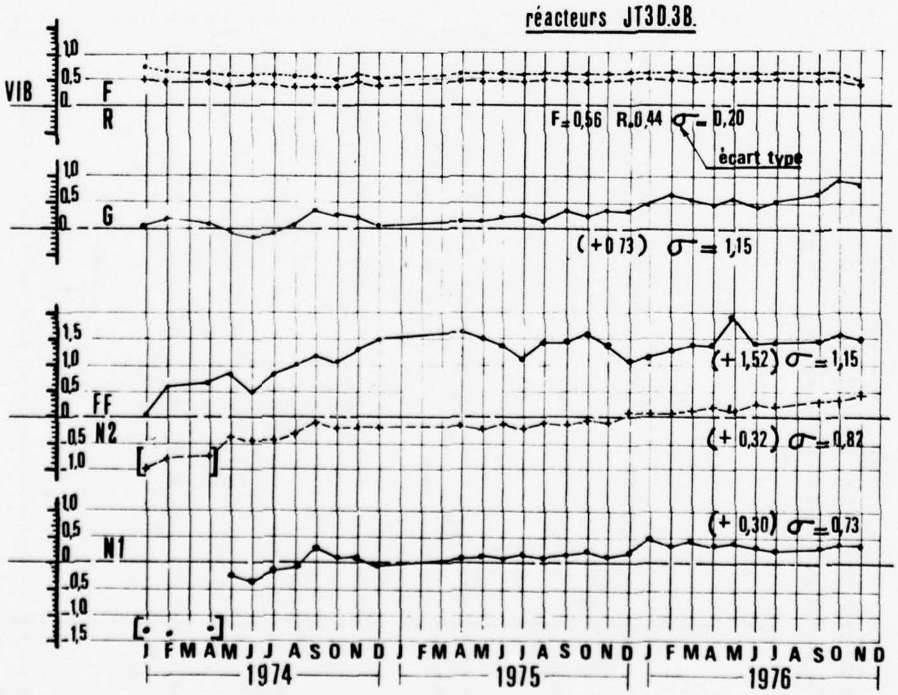
BTSC.3 -- CBE031



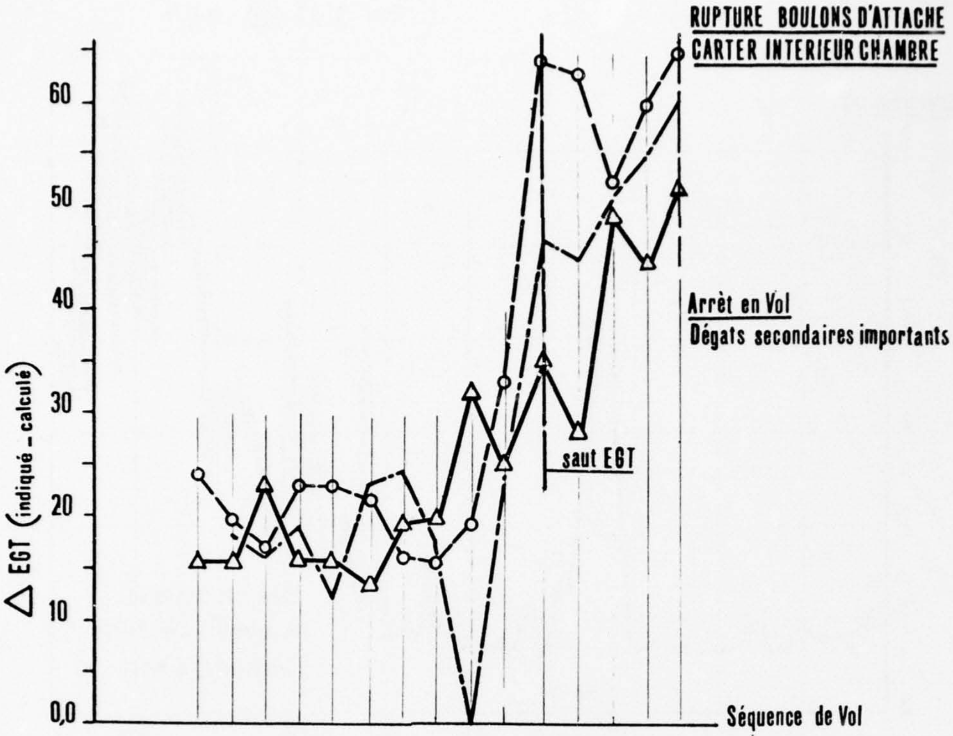
PROGRAMME_PWA FIG.8

Suivi des Performances moyennes
Flotte en croisière

réacteurs JT3D.3B.

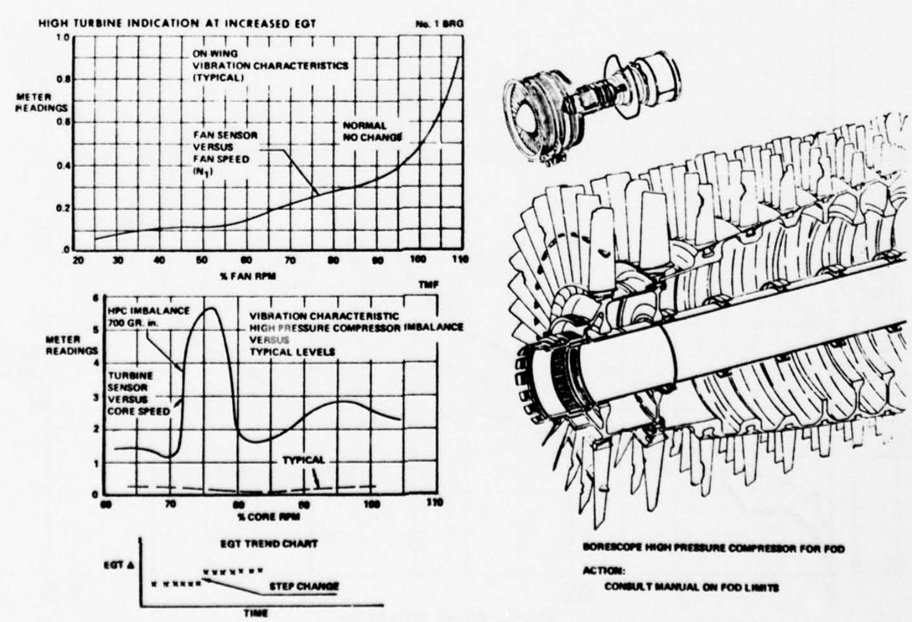


Réacteurs CF6.50 FIG.9

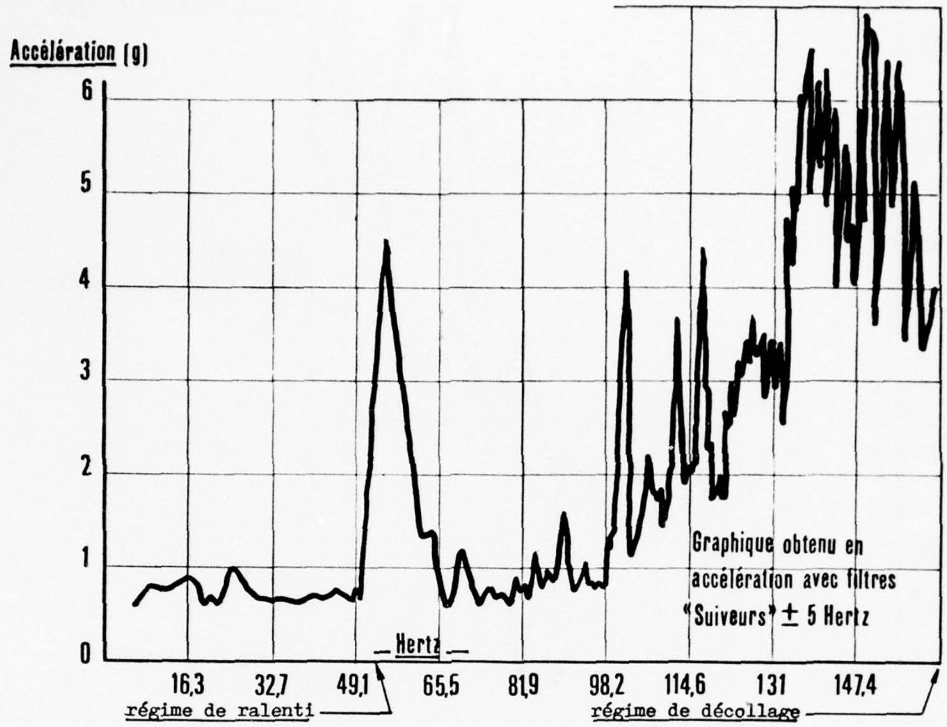


CF6.50 FIG.10

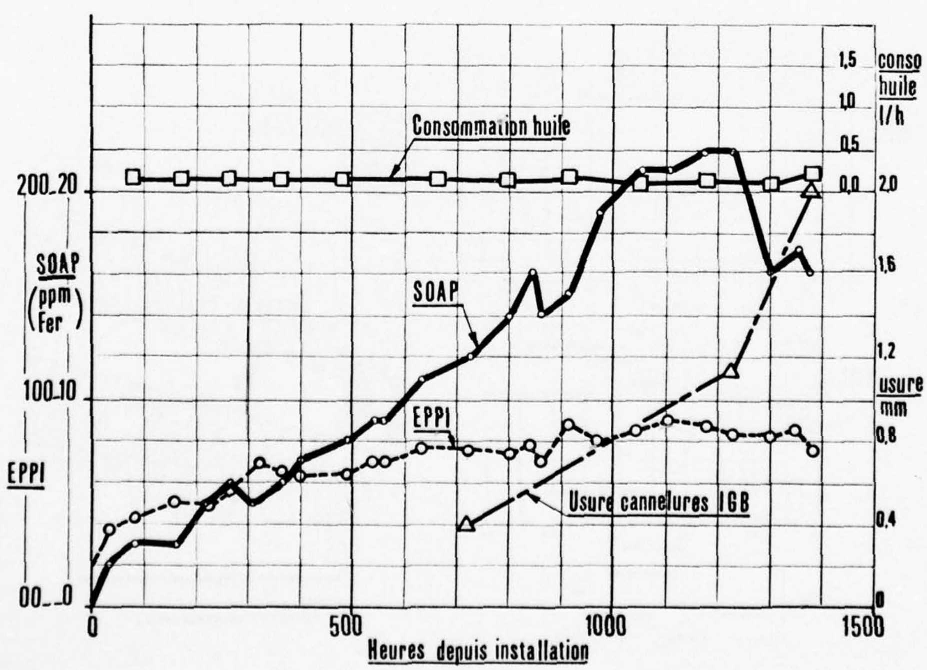
Manuel GE de Diagnostic de Vibrations



CF6-50 TMF Pick up FIG.11
réacteur n° 455400 14.11.75



DC-10 CF6-50 FIG-12
00SLB.1-455209 mars 1976



ATLAS CF6.50

BVGC.2 455706 _ 29.10.75

FIG.13

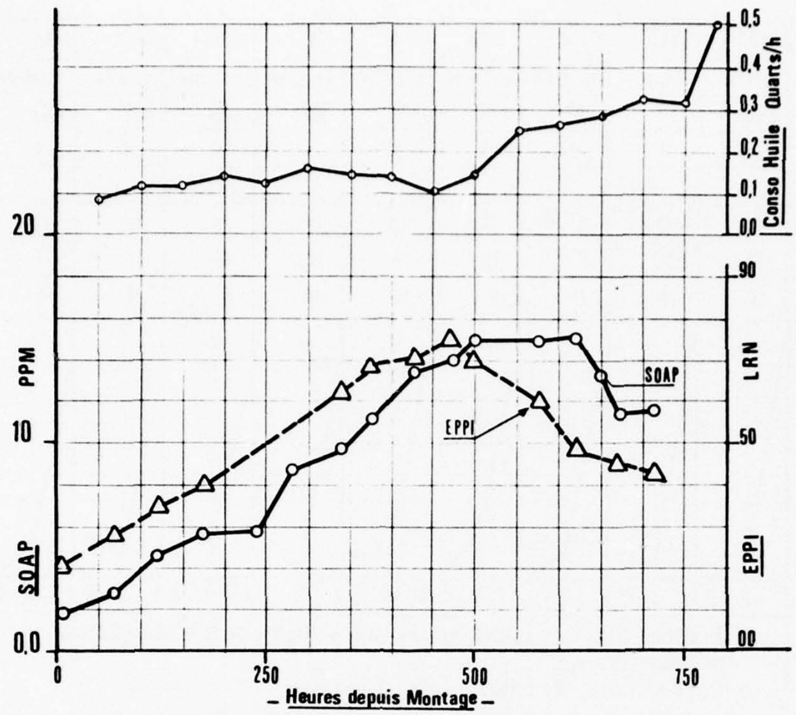
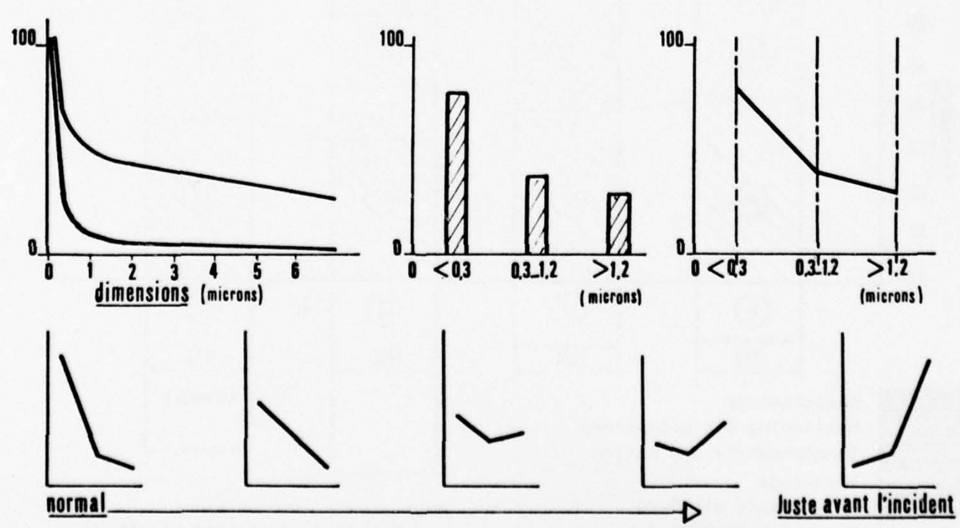


FIG.14

S.O.A.P et E.S.P

PROGRAMME D'ANALYSE D'HUILE PAR SPECTROGRAPHIE (Mobil)

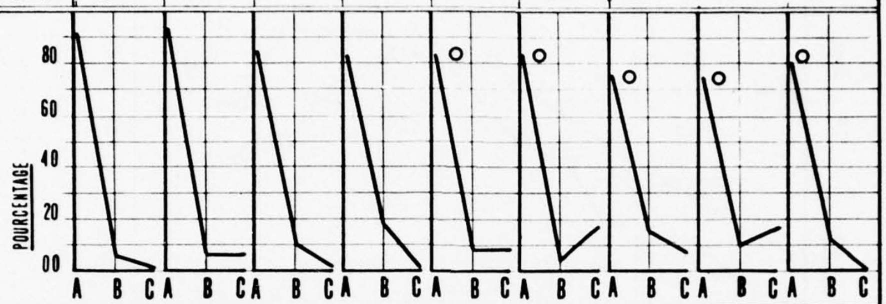
PRINCIPE
La répartition statistique en taille des particules de métal engendrées par ces phénomènes d'usure, varie de façon caractéristique lorsque le défaut s'accélère et que l'incident de fonctionnement de circuit d'huile est imminent.



Airline _ QANTAS/MAS
Engine n: 642682

FIG.15

Sample	7432	7507	7560	7625	7626	7707	7741	7757	140
Date	7.12.73	11.12.73	16.01.74	24.01.74	30.01.74	16.02.74	27.02.74	8.03.74	7.05.74
A/C Reg	EBS	EBS	EBS	EBS	EBS	EBS	EBS	EBS	EBS
Pos	4	4	4	4	4	4	4	4	4
T.S.O							18882		
Total Fe	20	22	8 △	11	12	12	12	13	22
Below 1,2 µ	20	21	8	11	11	10	11	12	21
Below 0.3 µ	19	20	7	9	10	10	9	10	18



Remarks

△ Oil change drops iron, however does not alter particle picture, which indicates no failure imminent

○ Small particles_ No failure in spite of high iron

ATLAS CF6_50

FIG.16

EFFICACITE RELATIVE DES DIFFERENTES METHODES DE MONITORING

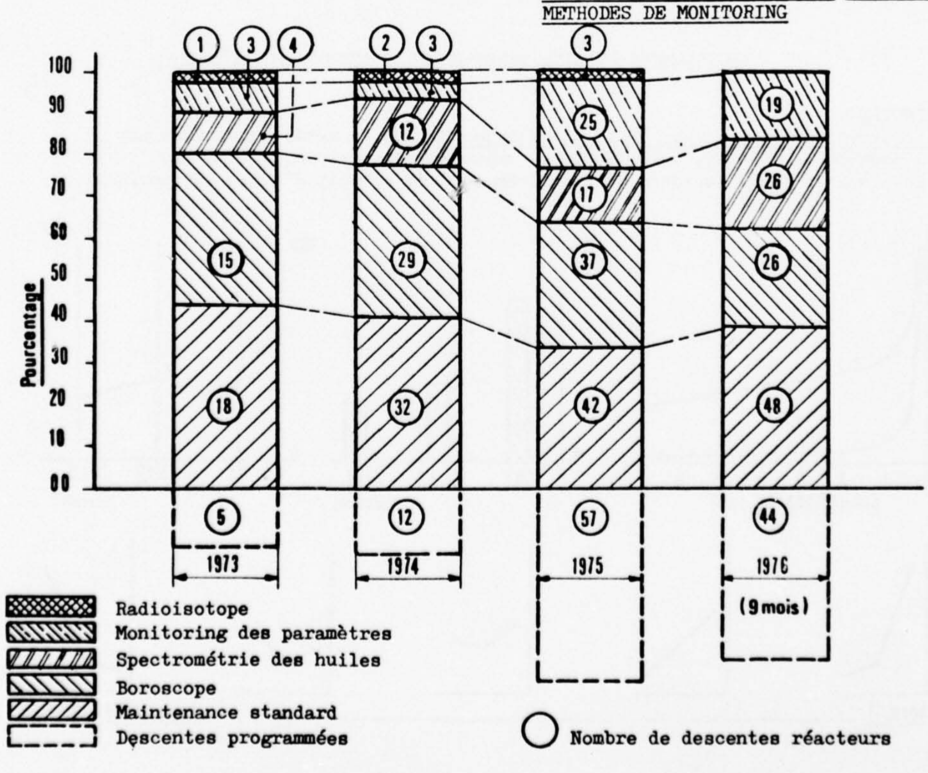


FIG. 17

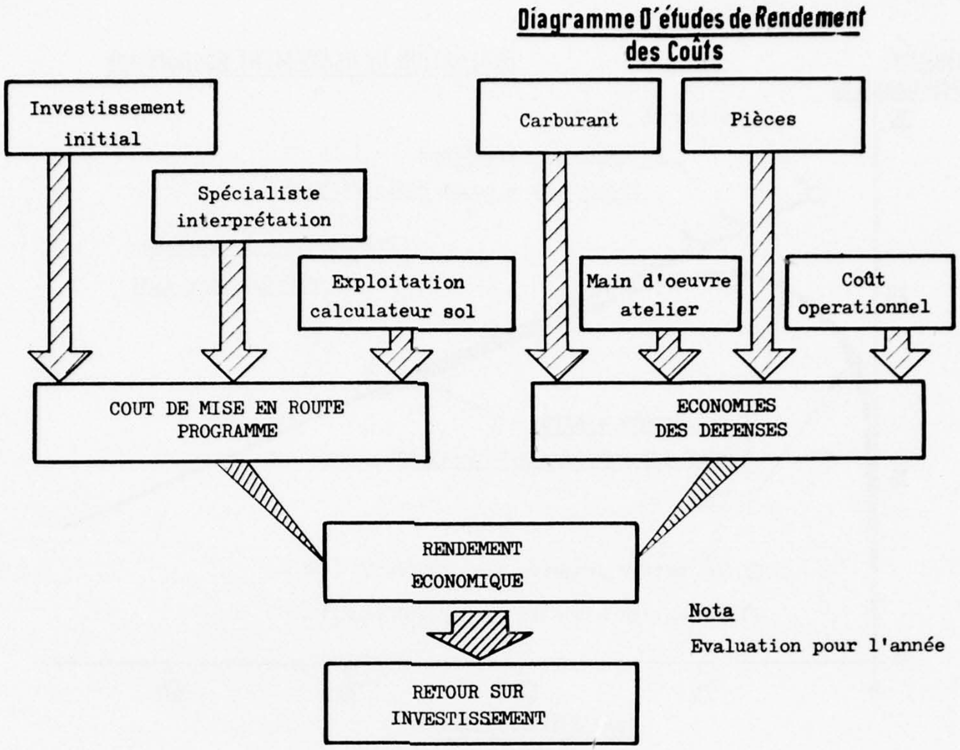


Diagramme D'études de Rendement des Coûts
Investissement initial
Spécialiste interprétation
Exploitation calculateur sol
Carburant
Pièces
Main d'oeuvre atelier
Coût operationnel
COUT DE MISE EN ROUTE PROGRAMME
ECONOMIES DES DEPENSES
RENDEMENT ECONOMIQUE
RETOUR SUR INVESTISSEMENT
Nota
Evaluation pour l'année

FIG.18

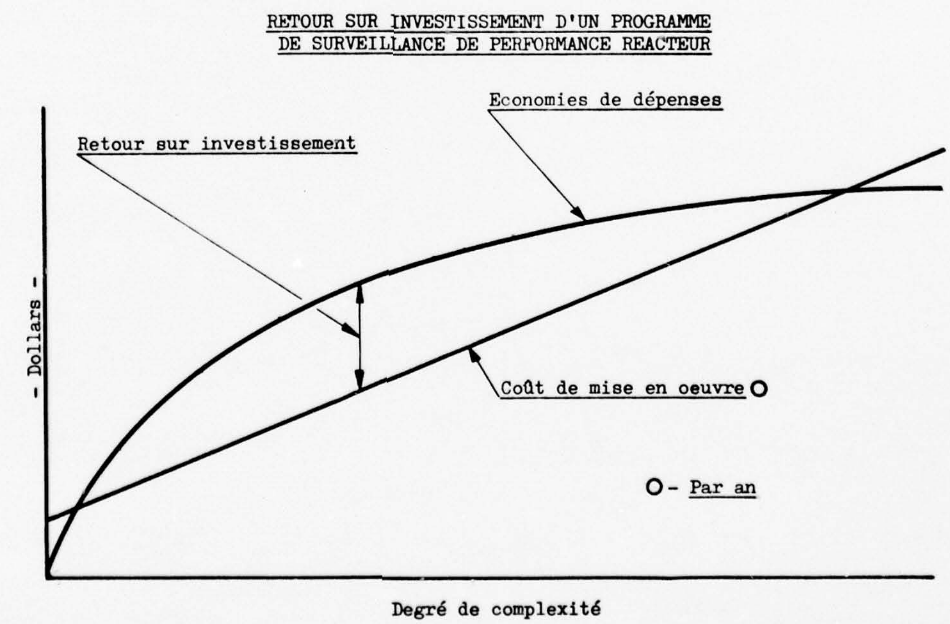
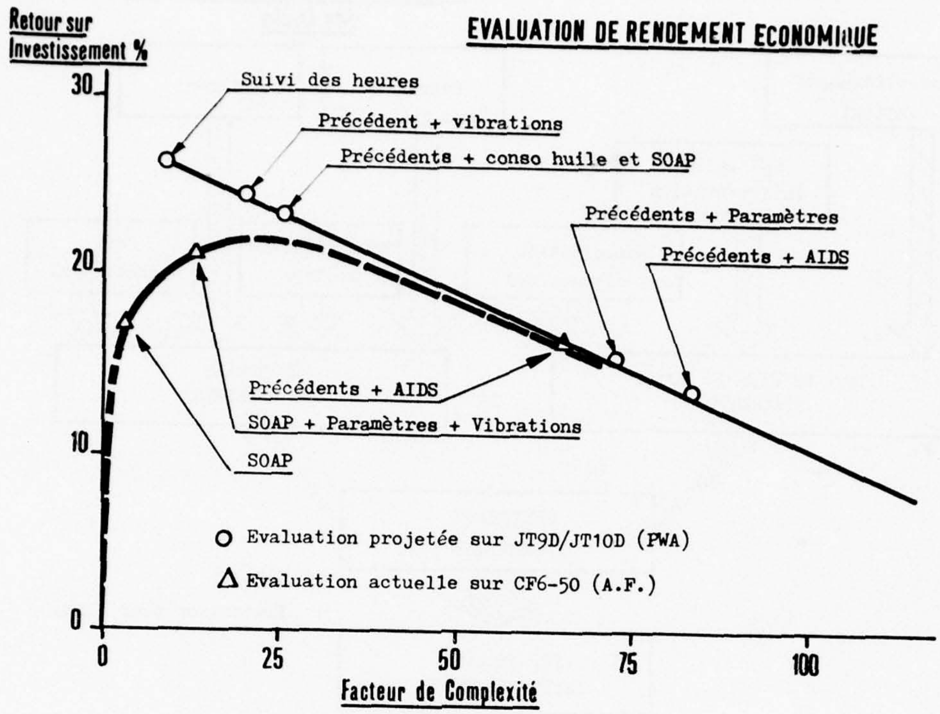

RETOUR SUR INVESTISSEMENT D'UN PROGRAMME DE SURVEILLANCE DE PERFORMANCE REACTEUR
Economies de dépenses
Retour sur investissement
- Dollars -
Coût de mise en oeuvre O
O - Par an
Degré de complexité

FIG.19



Retour sur Investissement %
EVALUATION DE RENDEMENT ECONOMIQUE
30
20
10
0
Suivi des heures
Précédent + vibrations
Précédents + conso huile et SOAP
Précédents + Paramètres
Précédents + AIDS
Précédents + AIDS
SOAP + Paramètres + Vibrations
SOAP
O Evaluation projetée sur JT9D/JT10D (PWA)
Δ Evaluation actuelle sur CF6-50 (A.F.)
0
25
50
75
100
Facteur de Complexité

DISCUSSION

R.Smyth

I would like to point out that Figure 19 showing investment returns against complexity is a basic law for the application of technology to increase the effectiveness of modern systems, e.g. a turbofan engine. The basic curve has a shape like the lower one for the CF6-50 and generally is a function of the year of application. This is connected with the standard of technology available at that time to introduce a certain factor of complexity. There are three aspects of this basic law for economic returns against complexity:

(1) Too low a factor of complexity will not be giving a high return of investment because optimum use of the technology available at that time is not being used.

(2) Too high a factor of complexity will prove to be unreliable and costly because the technology required is not yet developed enough at the time of application.

(3) There is an optimum for the economic returns at a certain factor of complexity for a certain year in which a definite standard of technology is available. In Figure 19 this is the peak of the curve at about 25% complexity factor. In the curve of technological progress this peak will be higher and to a larger factor of complexity.

Réponse d'auteur

Je suis tout à fait d'accord avec vous, pour compléter mon commentaire, je dirais que notre expérience à ATLAS montre que la courbe relative au JT9D est probablement largement différente de celle relative au CF6-50. Vous savez sans doute que le JT9D est un moteur sur lequel les méthodes d'analyses spectrographiques classiques sont impuissantes à déceler les défauts des parties huilées. Donc le "R.O.D. SOAP" est nul. L'efficacité des méthodes de suivi des paramètres sur le JT9D est maintenant largement démontrée et se situe probablement quelque part vers 20 pour cent en valeur "R.O.D.". Par contre, notamment parce que le JT9D est un moteur déjà plus ancien que le CF6-50, il apparait comme essentiel d'avoir des renseignements très complets sur les rendements modulaires pour optimiser le temps de retour en atelier, et il est vraisemblable que le point appelé en résumé "AIDS" permettrait seul apparament d'atteindre cet objectif et la courbe remonterait alors probablement assez haut.

J.A.Rowlands

(1) Do you see a difference in the value of various monitoring methods according to the average flight time? For instance, are complex methods of monitoring intrinsically more cost effective on short range high duty cycle operations when compared with one on long range-lon dwell/const. operational condition operations.

NOTE: Question is prompted by our need in the Military to evaluate cost effectiveness of engine health monitoring and flight data recording on short duration high intensity missions.

Réponse d'auteur

Votre question est tout à fait pertinente. Elle nous a conduit au sein du Groupe ATLAS à faire une étude exhaustive sur l'optimisation des méthodes de surveillance, en fonction justement de la durée du cycle. On s'aperçoit que les phénomènes mettant en jeux les phénomènes cycliques ont une influence relative directement fonction de la longueur du cycle: je pense à la résistance au fluage et à tout ce qui touche d'une manière générale les parties chaudes. Par contre, en ce qui concerne le rendement des parties froides (compresseurs) il semble qu'il soit d'avantage fonction du temps total, et donc, indépendant de la longueur du cycle.

TESTING SIMULATION OF DAMAGES OCCURRED IN SERVICE

by

D. Dini, University of Pisa, Italy
L. Giorgieri, Ministero Difesa Aeronautica, Italy

SUMMARY

This paper provides the basic framework from which further simulation of increased complexity and sophistication can be easily implemented on the field of engine failures by in flight foreign object, large overpressure signature inlet flow distortion, and icing environment.

The intention of this effort is to provide a general basic engine reliability program, capable of simulating a running turbojet-engine and its airsupplying environment as an integrated system, with the specific subroutines for the possible damages from foreign object and unsteady flow distortion, to be supplied by the user as required.

Recent advances on testing simulation of power plant damages occurred in service carried out at the University of Pisa promise to reduce accident causes, conditions and casualties, due to engine operation at low altitudes and in rugged confined terrain. This paper discusses state-of-the art design techniques to improve engine reliability and includes analysis of three particular experimental simulations to determine causes and effects and to make recommendations that will eliminate or reduce the causes of aviation accidents.

INTRODUCTION

The important features in this simulation are: acceptance of an engine normal rating operation, development of facilities for air breathing propulsion testing in all speed regimes, a detailed simulation of each abnormal dangerous inflow condition.

Engine reliability is challanged by the practical necessity of compromising inherent factors of design, environment, and operation. As aviation's role in public transportation has become firmly established, more and more attention has been devoted to ensuring the reliability, and therefore the safety, of flight in whatsoever environmental condition.

Operating experience over the years has indicated the need to consider in the mechanical design phase of an aircraft engine the possibility of ingestion solid objects including birds. These considerations are frequently based on fracture mechanics including estimates of impact energy absorption, and will usually include a subsequent experimental proofing phase. Other than to note that aerodynamic or other forms of damping are not significant in limiting the first peak excursion strains imposed by this type of loading, there will be no further consideration here of this problem; structurally safe foreign object ingestion is routinely provided by engine designers.

Nevertheless, it has to be recommended to inspect the visible parts of the engine inlet area for damage, to assure that the compressor turns freely before attempting a start, and to make a few quick routine checks for obvious failure. It is essential to inspect compressor stator inlet guide vanes, expecially the stages of variable vanes, and compressor rotor blades axially dovetailed into each rim and held in place by a retaining ring. If impact damages from foreign objects over compressor vanes and blades are realized, it is mandatory to reduce until acceptable size or to substitute each damaged vane and blade. Recommended minimal sizes are claimed by designers for check comparisons.

The foreign object debris spectrum, a typical engine can be expected to experience in its exposure to an uncontrolled operational environment, is quite broad. Experience with contemporary metals, such as stainless steel-and titanium, suggests that the degree of damage severity inflicted by these objects on the airfoil can vary considerably from a condition of localized deformation to complete material separation. The localized damage produced by small body impacts generally results in a reduction in fatigue strength while complete material or airfoil separation produced by larger body on blading result in either rotor unbalance requiring a reduction in engine speed or complete shutdown. Blade impact damage caused by small debris, such as aluminum or steel rivets, nuts or bolts, or stones and stack deposit from aircraft carriers, occurs most frequently while the aircraft is on the runway. Damage by hail and birds occurs mainly in flight. Primary impact damage due to foreign object particles occurs most frequently on the leading edge region or typical blading at location above the pitchline. During flight, the particle strike angle at the leading edge is strongly influenced by aircraft speed. Conditions governing angularity differences between the blading leading edge and foreign object particle may be analyzed, for instance the variation of particle impact incidence angle as function of relative aircraft speed on hub, pitch and tip sections, for a spectrum of engine blading.

A research like that has been carried out in our laboratory in Pisa on a T58 axial-flow turboshaft engine in a rig test facility, in which a perspective of the debris characteristics was obtained until reliable tolerance to sustain damage from F.O.D. into the compressor blading. Nearly 2000 horsepower was required to sustain compressor operation. Experiments were conducted in more phases, without replacement of damaged compressor blading, as long as localized deformations and material separations had enough mutual torque to stop the first stage rotor blades against the corresponding stator variable vanes. This test simulated efficiently an engine shutdown occurring for a small body (a rivet) ingestion.

Significant transient loading of compressor or fan structures are usually associated with a gross change in the throughflow that is truly aperiodic or of very low frequency. Examples may be related to compressor stall, surge, hammershock, bistable inlet operation, and air blast wave overpressure signature. The sudden changes in throughflow in treated experimentally, as in Ref. 1 regarding blast wave signatures. The problem is closely related to the aeroelastic behaviour of a single airfoil passing through a sharp-edge gust front in the atmosphere, such as might be produced by a remotely originated blast wave. However, the extent to which the elastic flexibility of a turbomachine structure contributes to, or interacts with, the distortion of the throughflow is at present a moot point. Simulation facilities, Ref. 1, are mandatory in treating the aeromechanical response of engine structure to transient loads.

Engine flameouts have been experienced while flying in high icing conditions. Testing simulation of ice ingestion in flight may determine the causes of engine flameouts and qualitatively assess the improvement to the intake duct anti-icing capability resulting from suitable devices.
As for the previously discussed F.O.D. and blast wave engine damage in service, testing simulation may be efficient means to investigate engine ice ingestion in flow.

In this paper results of experiments and proposals are presented.

TESTING SIMULATION OF FOREIGN OBJECT DAMAGE

Full scale static testing of actual inlet and engine installation has been conducted. The T58-GE-3 turboshaft for helicopters, Figure 1, producing 1250 shaft horsepower was chosen. The gas generator consisted of a 10-stage compressor, annular combustion and 2-stage turbine; the inlet stator vanes, Figures 2, 3, 4, 5 and 6, in stages 1, 2 and 3 were variable. The power turbine section consisted of a single stage turbine and exhaust section.

The relationship of aircraft velocity to F.O.D. velocity was derived on a relative basis from analysis of velocity vector for a spectrum of contemporary engine designs. Engine speed had minor influence on relative F.O.D. velocity. Increasing the aircraft velocity, the relative impact speed augmented only slightly between blade and foreign object particles, while with increasing relative air speed, hailstone impact velocity is affected considerably.

In the engine under testing, a variable stator system solves the stall problem at intermediate and low speeds and permits the use of a single-spool high-ratio compressor. The variable vanes are nearer the closed position during engine starting; they open as engine speed increases. Varying the stator vane angle changes the direction and magnitude of the air velocity vectors, leaving the stator vanes so that the proper angle of attack is preserved with respect to the rotor blades.

Impact location and damage of blading was sufficiently consistent for most cases to determine fatigue strength of blades damaged by a prescribed mode of foreign object damage. Examination of the blades strain distribution revealed that the airfoil pitchline location experienced 60% of the maximum blade stress and therefore possessed the advantage of being located in an area of sufficiently high dynamic stress to induce failure in the impacted area. Experimentation was carried out to establish the fatigue strength relationship of undamaged blades to that of the previously impact damaged blades, electronically measuring by an electromagnetic counter the number of cycles to failure on a magnetic shake table. A baseline of undamaged fatigue strength was established initially for comparison to the damaged blades.

Blade impact from small bodies, as sand and metal shaving with size under 1/10", caused limited damages on the compressor blading. The residual dye penetrant inspection fluid revealed crack formation on the trailing edges; but, in general, only diffused abrasion on blade surface was detected.

From the results of these tests, it was concluded that the shear-type impact could be reproduced reliably using aluminum balls as projectiles, passing through the air inlet rhomboidal wired grid in front of the engine.

Systematic tests were carried out on the T58 turbojet engine. After fatigue strenght evaluation as consequence of airflow mixed with fine sand particles, the blade leading edges were submitted to heavier ingestion from which resulted indented and deflected by metal shaving suspended in the airflow entering the wired grid.

Foreign object ingestion testing was continued as long as the forward deflection of the tip leading edge of some first rotor blades was increased almost to touch the first stator vanes.

The twisted and distorted first rotor blades were not substituted, so that, as consequence of a larger aluminum body ingestion passed through the wired grid, blades and vanes of the first stage impacted one another so strongly to stop entirely the engine. As it will be seen later in the pictures, the small ob-

ject was restrained in between tip vanes and blades, in such way to determine the forward twisting of a rotor blade which impacted the stator vanes after the rotor blade fixing ring Table VI failure.

This fact is showing that small size foreign objects entering the standard airflow filtering wired grid may cause engine stopping, if blade inspection is not accurately carried out before each flight, eliminating strains and distortions by blade size reduction until minimal acceptable standard or substituting the damaged blades.

The compressor rotor is shown in Figure 7.

Tables from I to XIII show the simulation effect produced by F.O.D., in such degree to stop at all the engine in flight. Only a few pictures of the complete macro and micro documentation are presented.

Table I is regarding the first rotor blading strains in the above mentioned stopping condition. For comparison, Table II presents the quite good condition of the second rotor blading, which revealed only abrasions.

The engine stopping by F.O.D. simulation may be analyzed through the following Tables:

- Table III - first stage vanes and blades;
- Table IV - first rotor blades;
- Table V - first stage variable vanes;
- Table VI - first rotor blade and fixing ring (new and undamaged);
- Table VII - examples of first stage damaged blades and vanes;
- Table VIII - mutual position of new and damaged vanes and blades of the first stage, with blade fixing ring released;
- Table IX - microphotographs of blade number 10, showing strains determined in three successive F.O.D. phases;
- Table X - macro and microphotographs of blade number 1, from which it was possible to rebuild the story of the previous and last strains;
- Table XI - old and new strains in blade number 2;
- Table XII - blade number 24 (4 pictures); with yielding trace; blade number 25 (2 pictures); blade number 26, reduced in size before the last running test;
- Table XIII - typical simulated impact damages.

The present F.O.D. simulation, limited to compact turbojet engines with small blading clearances, is confirming the necessity of frequent compressor blade inspection, elimination of distorted parts and substitution of more damaged blades.

A small foreign object, less in size than the filtering grid holes, may severely damage already distorted blades, even until stator and rotor impact and engine stopping.

According the described test, only the compressor resulted damaged. The first stage determined rapidly the engine stopping. The last high compression stages had all blades and vanes distorted by minute metal particles. No impact traces were observed inside the compressor casing, as well as in all parts of the compressor and power turbines.

The microscopic investigation permitted to detect, on the distorted blades, more phases of deformations occurred in different running tests, as result of fatigue phenomena.

The simulation rig for F.O.D. investigation has required shielding and protection against possible compressor casing explosion.

TESTING SIMULATION OF STRUCTURE DAMAGE FROM TRANSIENT LOADS

Engine acceleration and deceleration can be considered to be quasi-steady because their time is typically from 1 to 10 seconds, wehereas the transport time of particles through the machine is a few milliseconds. Transport phenomena relating to flutter, resonance, inflow distortion, aerodynamic stability, surge and rotating stall, are basically unsteady requiring unsteady analysis.

So, aeromechanical response and possible damage in turbomachinery may be analyzed by recognizing experimentally the unsteady nature of the inflow as consequence of overpressure signatures produced by a N-wave simulator.

The entire phenomenon of the aerodynamic moment resulting from aperiodic forcing, and the resulting torsional and bending blade or vane response, because of short duration strong flow distortions, may be simulated with an open end shock tube, Ref. 1. The explosive charge at the closed end causes a shock front to reach the open end, simulating the effects of very larger detonating spherical charge. In a series of tests with the CAMEN shock tube, Figure 8 and Ref. 1, a turbojet engine was operating outside, close the open end of the tube, for simulation of the effect of sonic bang or similar inflow transient distortion.

This simulation facility is capable of varying pressure-time diagrams in a predictable and simple manner, providing appropriate frequency and velocity of the travelling wave. Comparing the N-waveform of a typical sonic bang with the waveform of the bang from the explosion of a point source charge, one sees that there is clearly more "energy" in a sonic bang than in an explosive bang of the same pressure rise across the shock, owing to its longer duration, and hence to its greater impulse. Even though the waveform of the bang simulated by two point charges delayed relative to the other by an interval of time may be a good simulation of sonic bang or similar transient load, it is preferred an explosive line charge formed, by superposition, with a number of strands whose ends are successively offset relative to the others. In this way it will be possible to construct a charge whose waveform approximates that of the desired N-wave or other overpressure signature.
Simulation of upstream transient turbulence or load entering an engine inlet, quite strong, for instance, at high flight Mach number and at small distance of the acoustic spreading supersonic plane, can be approached in several ways.

Compressor response to overpressure loading, generated by a linear explosive charge formed by superposed strands, was observed with the CAMEN shock tube facility. Mass flow change, angular velocity decrease, and axial thrust against the compressor, resulted acceptable for average overpressure of 70 psf, whereas flow and compressor blading were seriously influenced by higher average overpressure.
It has been observed that the sudden change in the throughflow is associated with significant transient aeroelastic loading of compressor or fan structure. The most significant aeromechanical response was the periodic forcing experienced by rotor blades. Periodic aerodynamic and inertia blade loading had serious consequences with respect to the discs and shafts to which these blades were attached. Flutter with either random or uniform phasing between adjacent blades exerted oscillatory root reactions. Operating in highly unsteady throughflow, massive rotating assemblies with gyroscopic forces, friction, trapped fluid, compliant bearings, etc., showed a rich variety of vibration phenomena.
However, transient aperiodic disturbances under 70 psf of overpressure were non important, since the vibration soon damped out due to aerodynamic and other forms of damping. Operating in a cyclic surging or stalling condition, the fatigue number of cycles ammassed in a short period of time brought to a rotationally symmetric excitation imposing axial loads on rotors and shafts. These structures, very stiff in the axial direction, with exceedingly high natural frequencies in their longitudinal modes, usually showed quite safe response.

Testing simulation of turbojet engine structure damage from transient loads, to improve reliability in service, may be performed in wind tunnels or large open end gun shock tube.

TESTING SIMULATION OF ICING ENVIRONMENT IN HELICOPTER FLIGHT

One recent study, Ref. 2, showed that 39.2 percent of all U.S. Army helicopter accidents occur during autorotation, as consequence of engine shutdown. For that, future helicopters will be equipped with crashworthy fuel systems complete with self-sealing, breakway valves and fittings, and fuel cells withstanding a drop test from a height of 65 feet onto a non-deforming surface without a leak. From the other hand, test programs are in progress to determine under which situations the crews could safely bailout a helicopter in various autorotation attitudes.

One of the most recurrent power plant failure is engine flameout while flying in light icing conditions, very important to be reduced in as much as accident rates (number of accidents per 100,000 flying hours) for helicopters is higher than for fixed-wing aircraft.

A number of icing projects have been performed in the past years to carry out an investigation on engine flameouts in light icing conditions, using static icing rig to improve engine intake duct.
Special instrumentation was installed in the aircraft, by Aerospace Engineering Test Establishment in Canada, to record with a visicorder and a camera the following engine parameters: inlet guide vane position, inlet temperature, maximum pressure, torque, propeller rpm, turbine temperature, turbine rpm percent, and fuel flow. The objectives of this program were to attempt to determine the cause of engine flameouts in light icing conditions, and to assess qualitatively the improvement to the intake duct anti-icing capability. Icing conditions were varied from test run to test run, but experience indicated that, for the ambient conditions available, a liquid water content of 0.20 gm/m^3 to 0.35 gm/m^3 and a mean droplet diameter of 20μm yelded the best results. The icing cloud was wide enough to completely cover the engine at stand point. Several test runs were attempted with all anti-icing and de-icing systems and ambient air temperatures from - 2.2°C to -11.6°C. Some test runs resulted in compressor stalls, or engine flameouts, or reactions to the throttle inputs.

The essential need of a european icing test facility for helicopter in-flight has been discussed. A specification of the proposed facility was defined; but the work aimed at designing such a tunnel was split up between different options, and until now no preferred option was selected. Previous recommendations to carry out engineering studies have been taken up and led to worthwhile activities and valuable results.

The requirement and specification for helicopter in-flight test facility are, respectively, determined and developed. All first line helicopters should have to be tested, and anti-deice equipment for helicopters should be annually requested. Main objectives are those to determine the capability of helicopters to safely operate in an icing environment, what anti-deicing equipment is needed on helicopters to fly into

known icing conditions, and flight envelope restrictions.

The proposal is to build a test rig in which an artificial cloud, formed from a large number of nozzle mounted on a rectangular grid, is blow over a hovering helicopter by a bank or banks of fans. At the test point, 100 feet from the grid, the cloud will be ideally some 100 feet wide by 15 feet deep with its centre line 80 feet above the ground. The air stream will be sufficiently large to maintain a reasonable consistent flow throughout the whole area of the cloud. In particular the lower edge of the cloud must be clear cut and should be maintained almost horizontal for at least 65 feet from the grid. A mean air speed of 10 knots is to be achieved at a distance of 100 feet from the fans, with a local variation of velocity non in excess of ± 1.5 knots within a circle of diameter 65 feet, with its centre 100 feet from the fan (or at distance of 65 feet, *with excess velocity of* ± 0.5 knots within the same circle). The overall velocity variation within the above circle is not to exceed, respectively for distance 100 and 65 feet from the fans, ± 3 and ± 2 knots longitudinally, and ± 1.5 and ± 0.75 knots laterally.
The rotation imparted to the air by fans will have to be overcome, possibly by controrotating fan units or by flow straightners in ducts or by a combination of the two. Multiple jet (supplied by indipendent blowers) confined mixing entails the transfer of kinetic energy from an array of primary stream to a secondary stream enveloping them.

The intent of this proposal is to stimulate advances in the area of helicopter testing techniques for better reliability in service.

CONCLUSIONS

F.O.D. simulation, as carried out on a used engine, has shown that it is mandatory to substitute damaged compressor blading, or at least to eliminate each small strain, in order to avoid possible engine stopping as result of further F.O.D. entering the standard wired grid.
In the present case, already strained (as consequence of previous limited F.O.D.) first stage blades and vanes impacted one another because of a 1/10" diameter alluminum rivet entered the standard wired grid. This rivet, entering at the tip of the compressor first stage in between variable vanes and blades, was able to twist and deflect a blade as long as the blade sealing ring became open. The blade itself moved forward and impacted the variable vanes. Only compressor blading resulted damaged.
The accurate investigation of damaged blades and vanes has permitted to detect all the deformations produced by the previously entered foreign objects. In particular, the tip blade twisted sheets resulted as consequence of successive superposition of F.O.D. running tests.

Gun shock tube may be efficiently used as simulation device for strong and short duration airflow unsteadyness in actual size turbomachinery.

ACKNOWLEDGEMENTS

The authors wish to gratefull acknowledge the contribution made by Mr. A. Cecchetti, graduate student at the University of Pisa, in carrying out the macro and microphotographs.

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2) J.L. Haley, "Analysis of US Army Helicopter Accidents to Define Impact Injury Problems", AGARD Proceedings No. 88 on Linear Acceleration of Impact Type, June 1971.

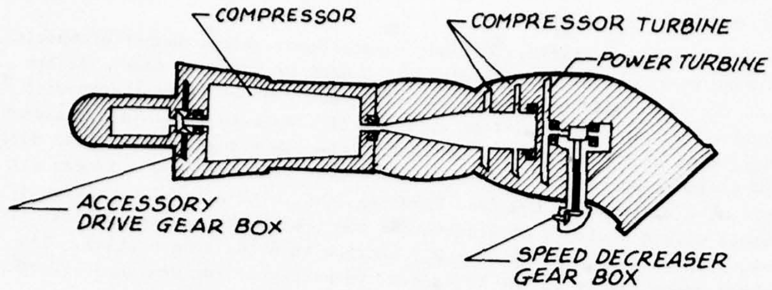

Fig. 1 - T58-GE-3 turboshaft

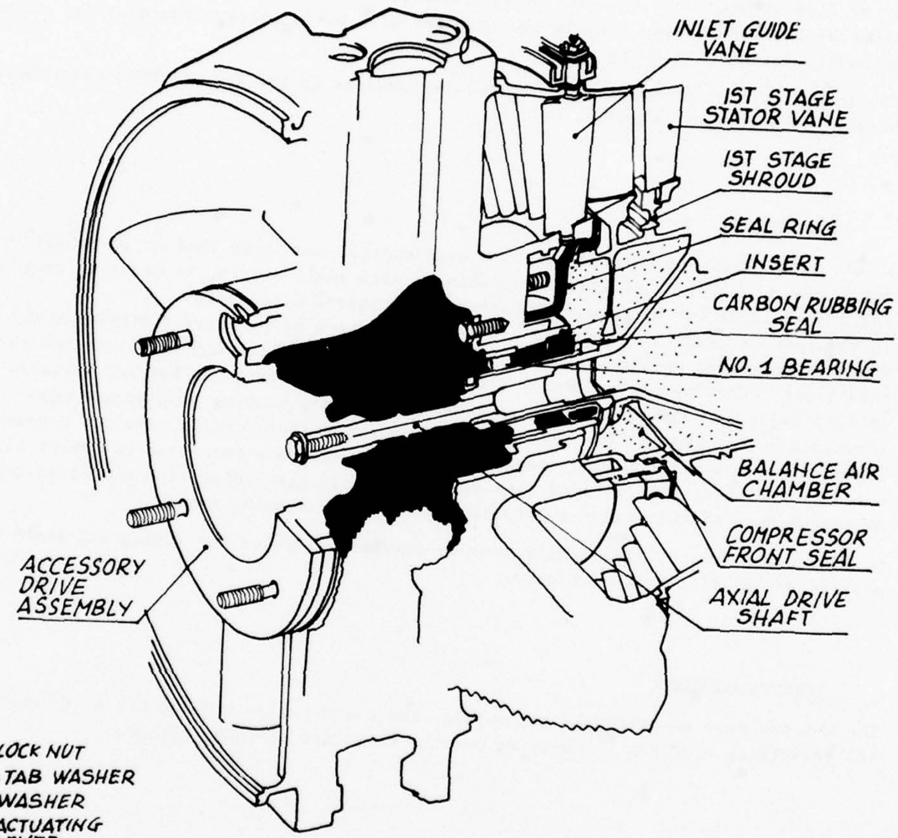

Fig. 2 - Engine intake and compressor first stage

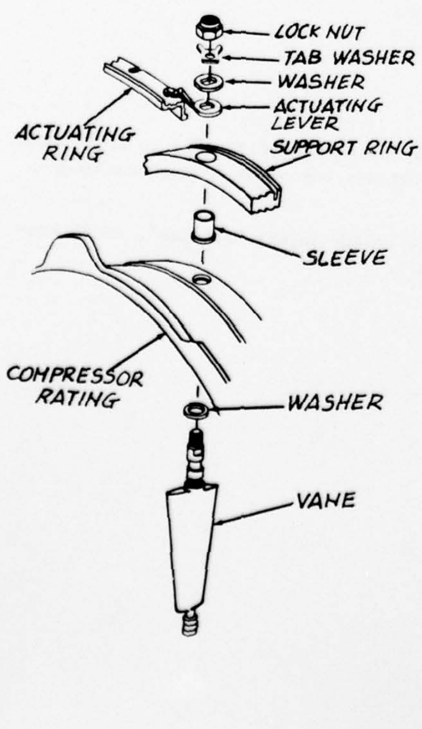

Fig. 3 - Variable inlet guide vane

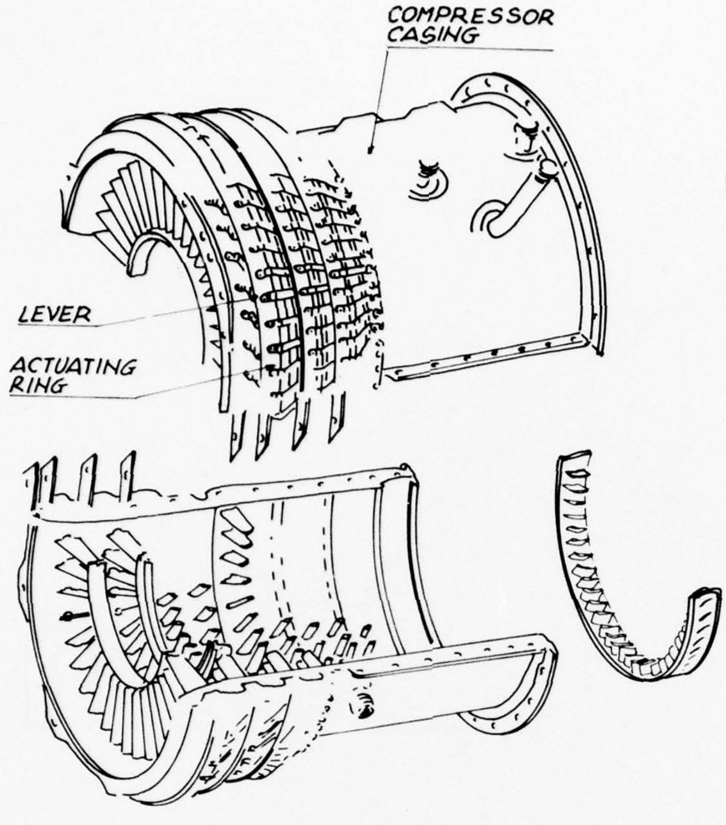

Fig. 4 - Compressor casing with variable vane actuating ring

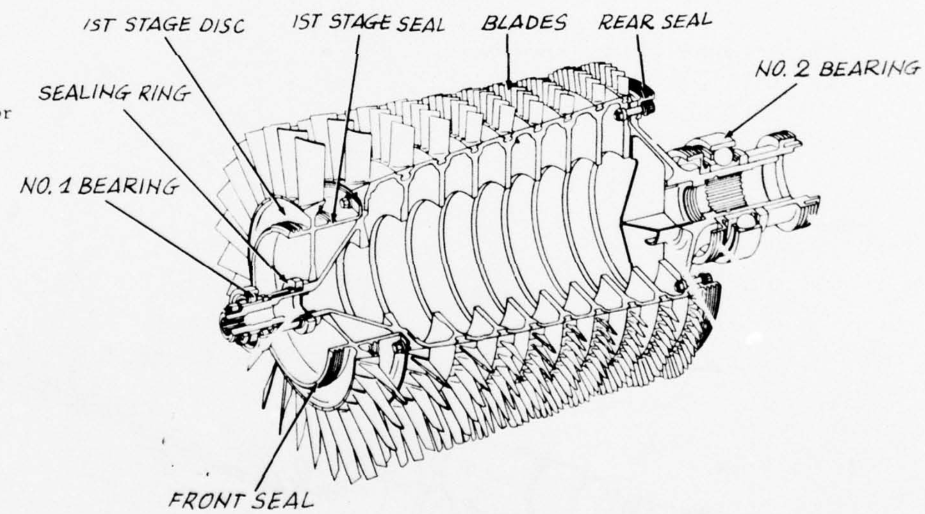

Fig. 5 - Compressor rotor

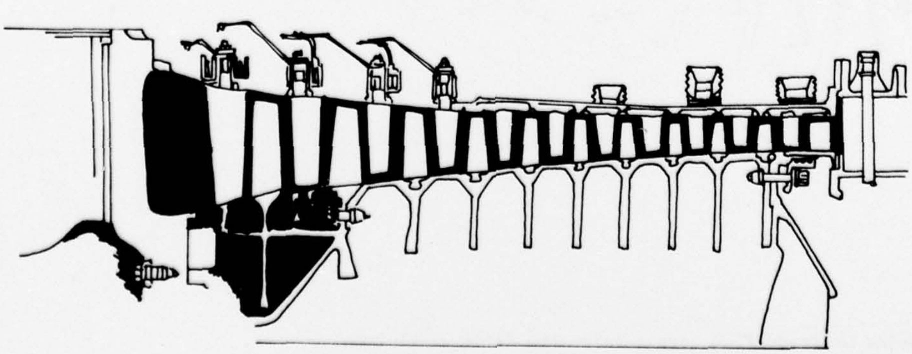

Fig. 6 - Compressor blading: variable and fixed vanes, and blades

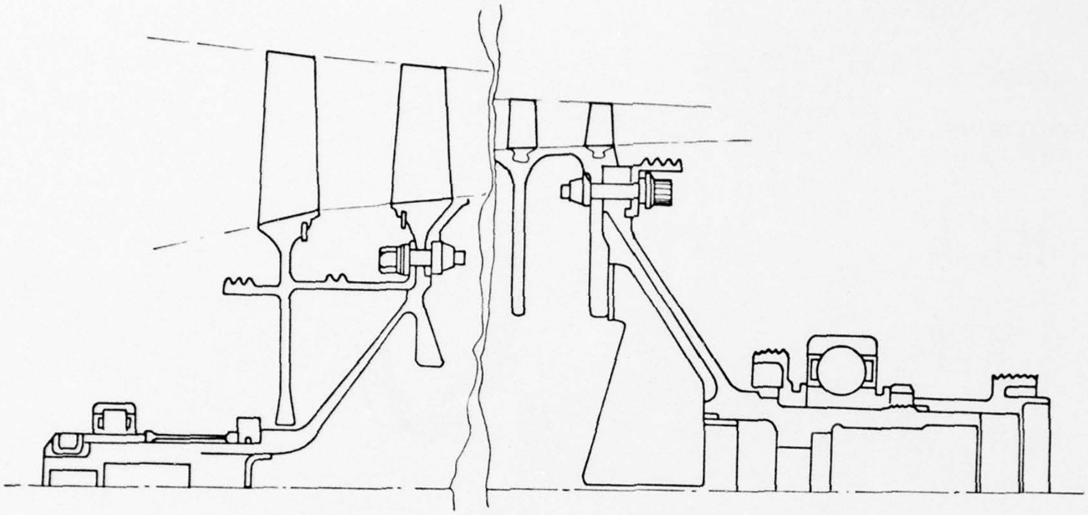

Fig. 7 - Rotor assembly compressor

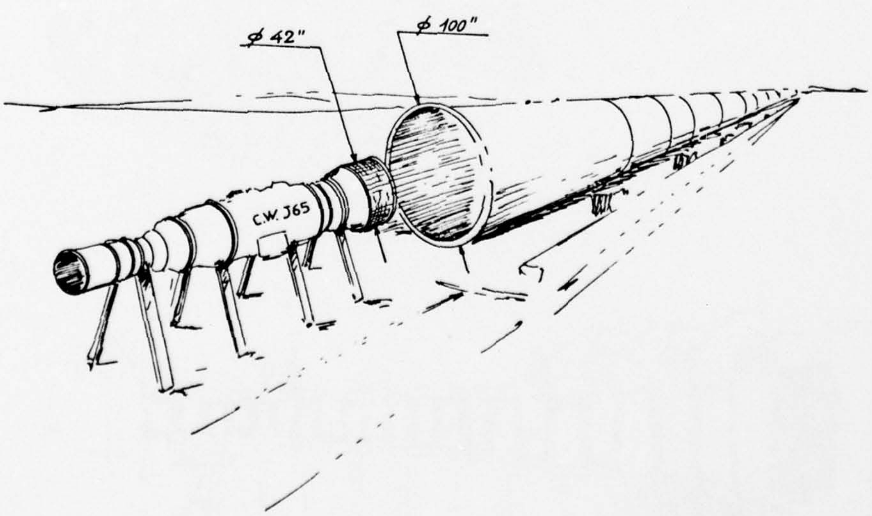

Fig. 8 - Turbojet unsteady flow test outside the CAMEN shock tube

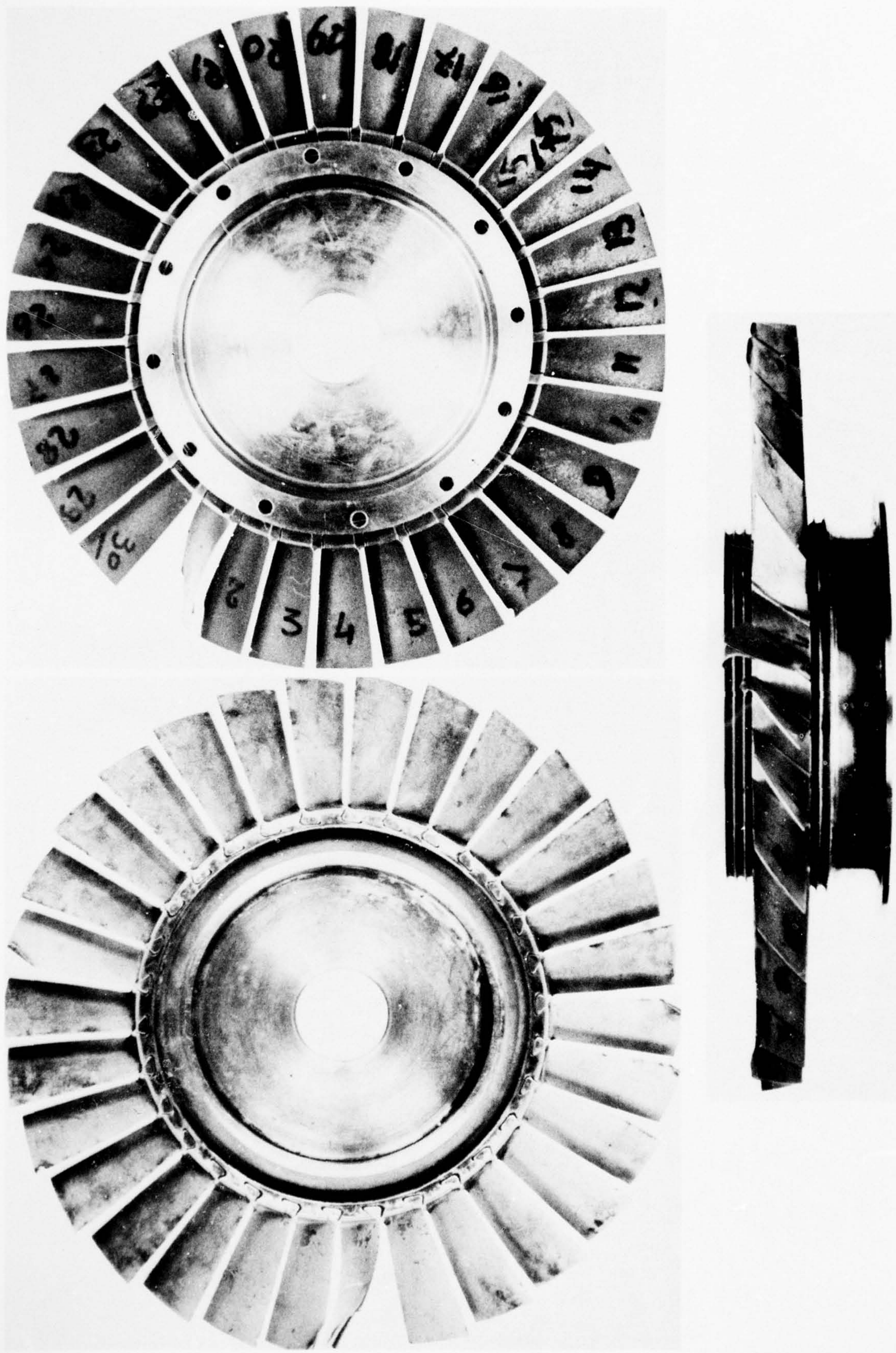

Table I - First rotor blading strains

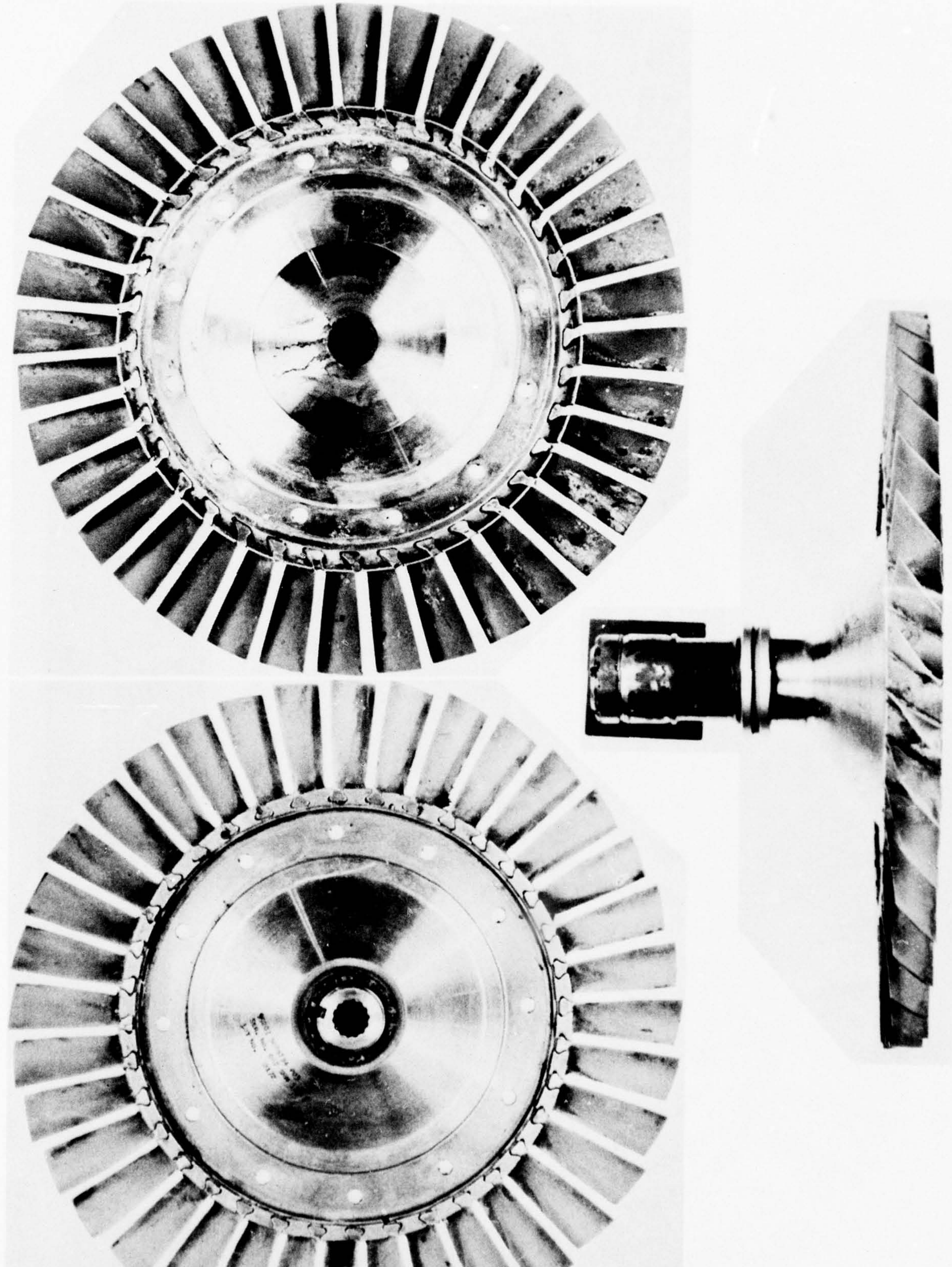

Table II - Second rotor blading

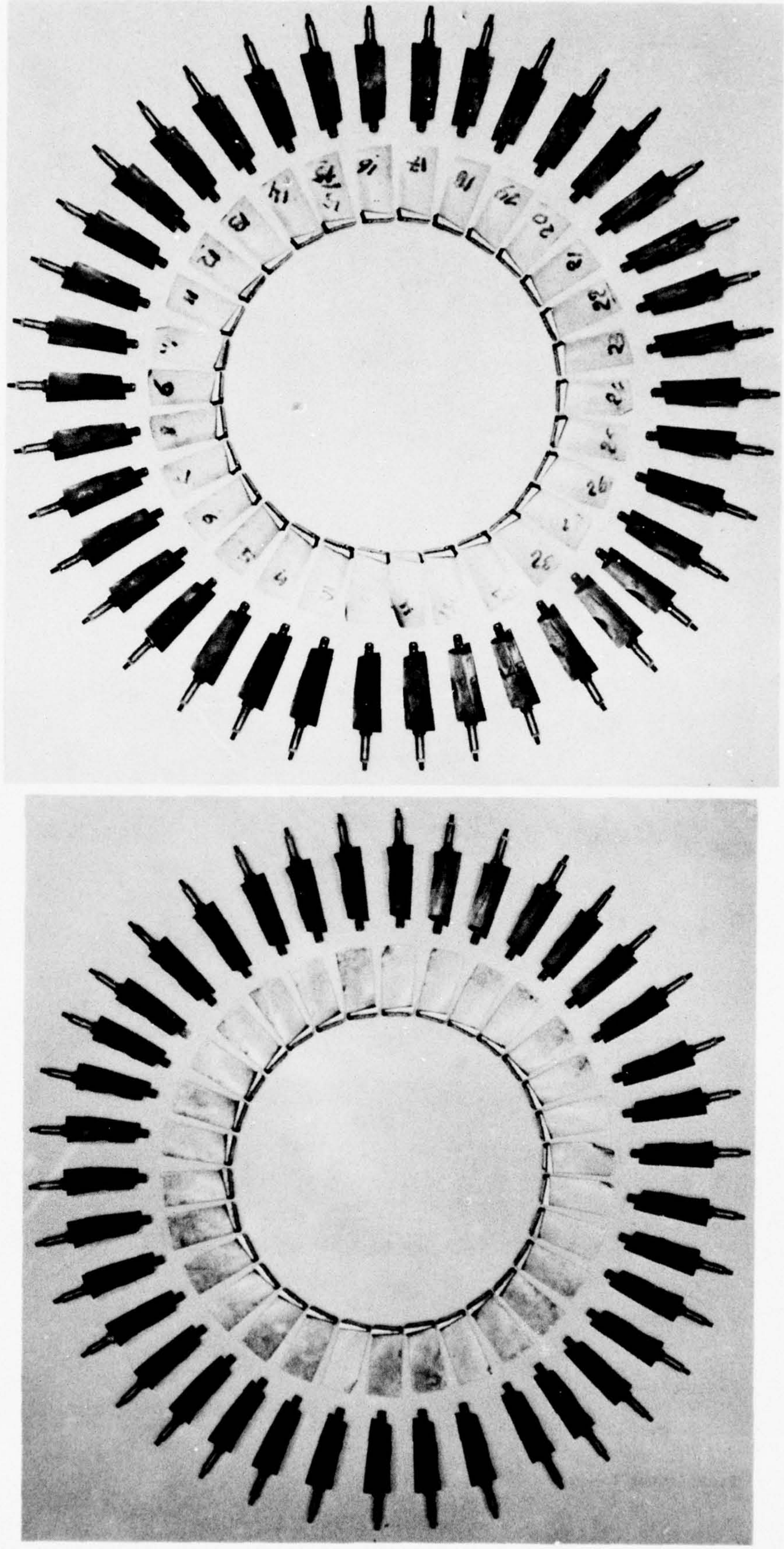

Table III - First stage vanes and blades

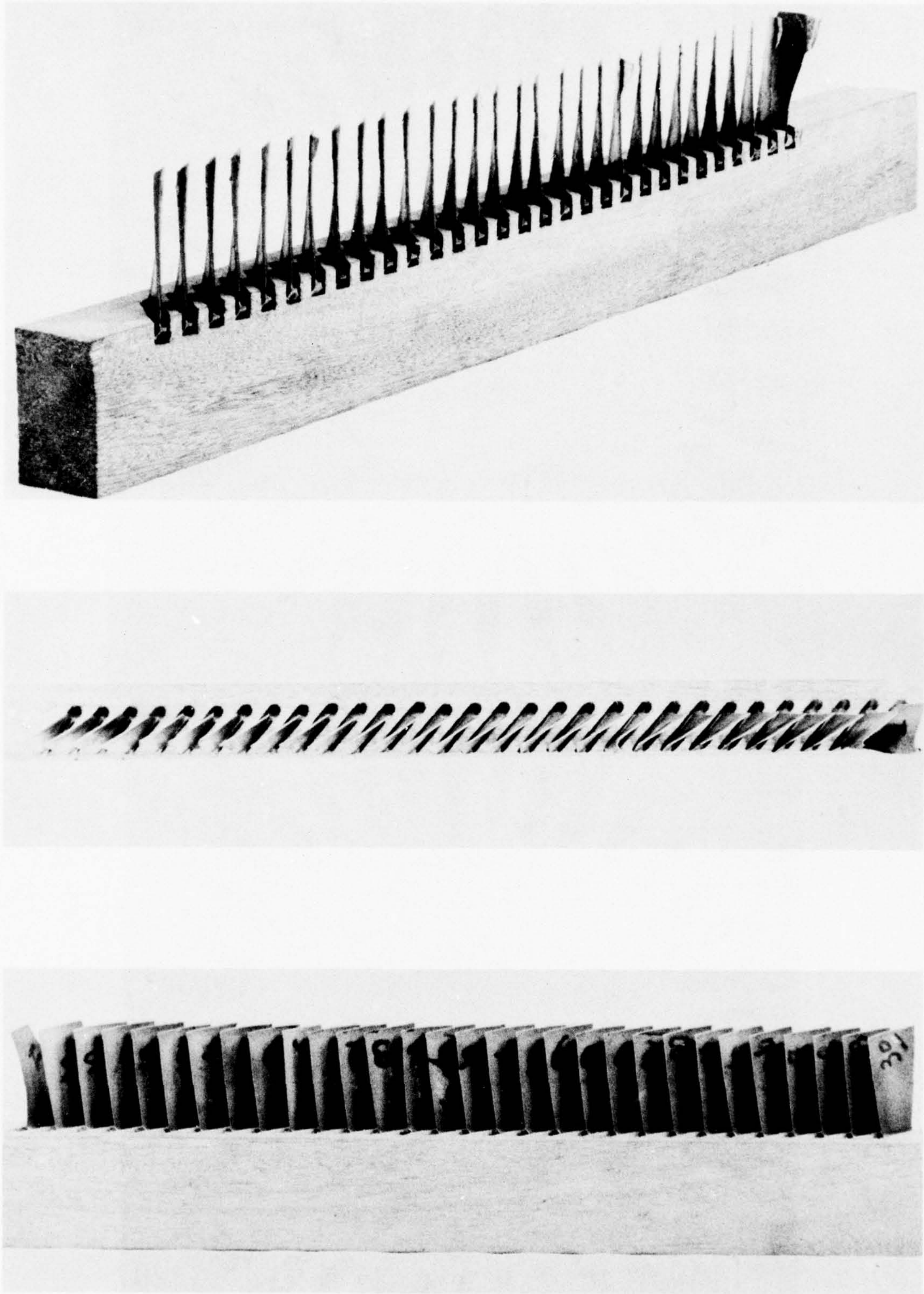

Table IV - First rotor blades

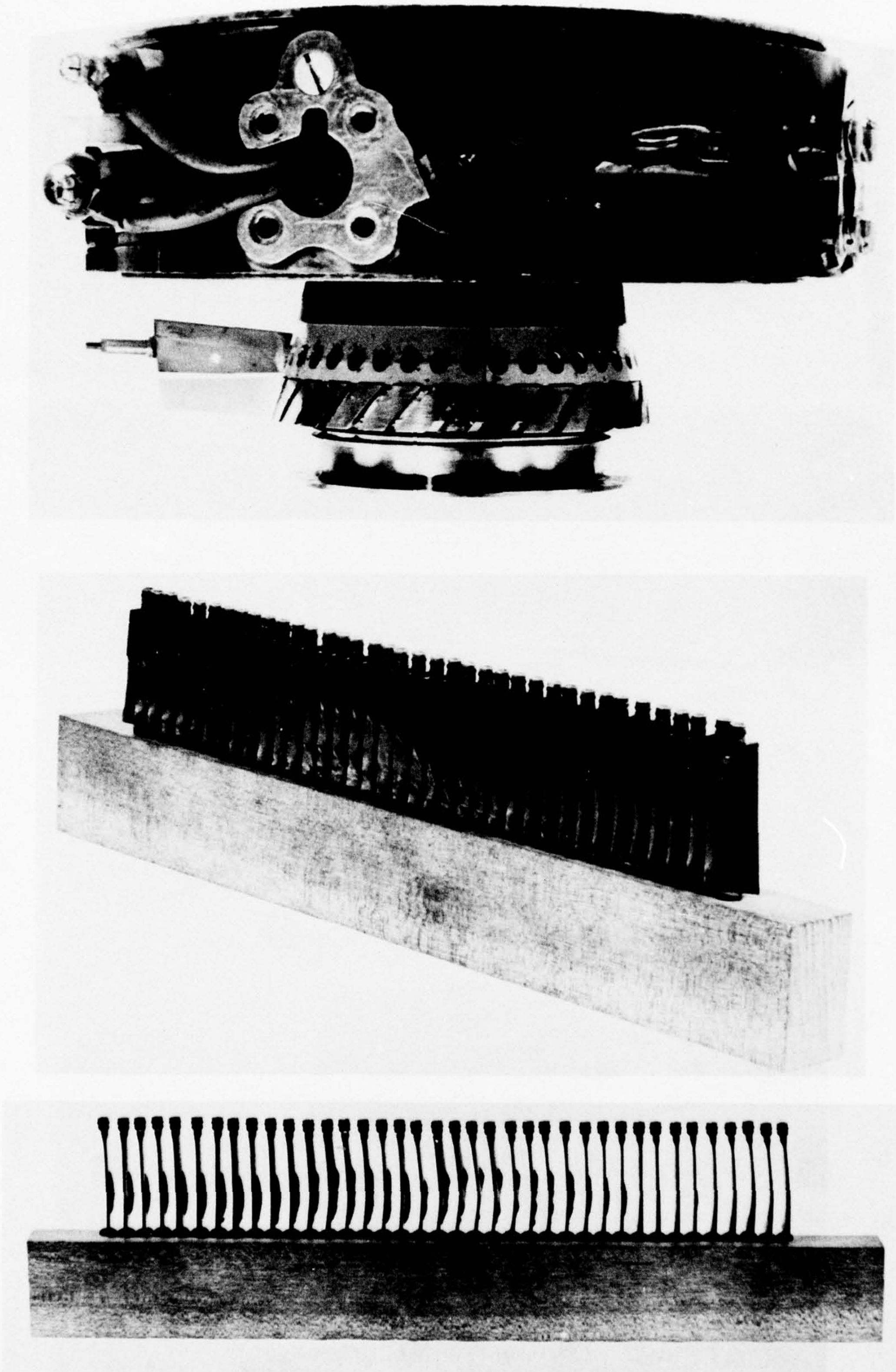

Table V - First stage variable vanes

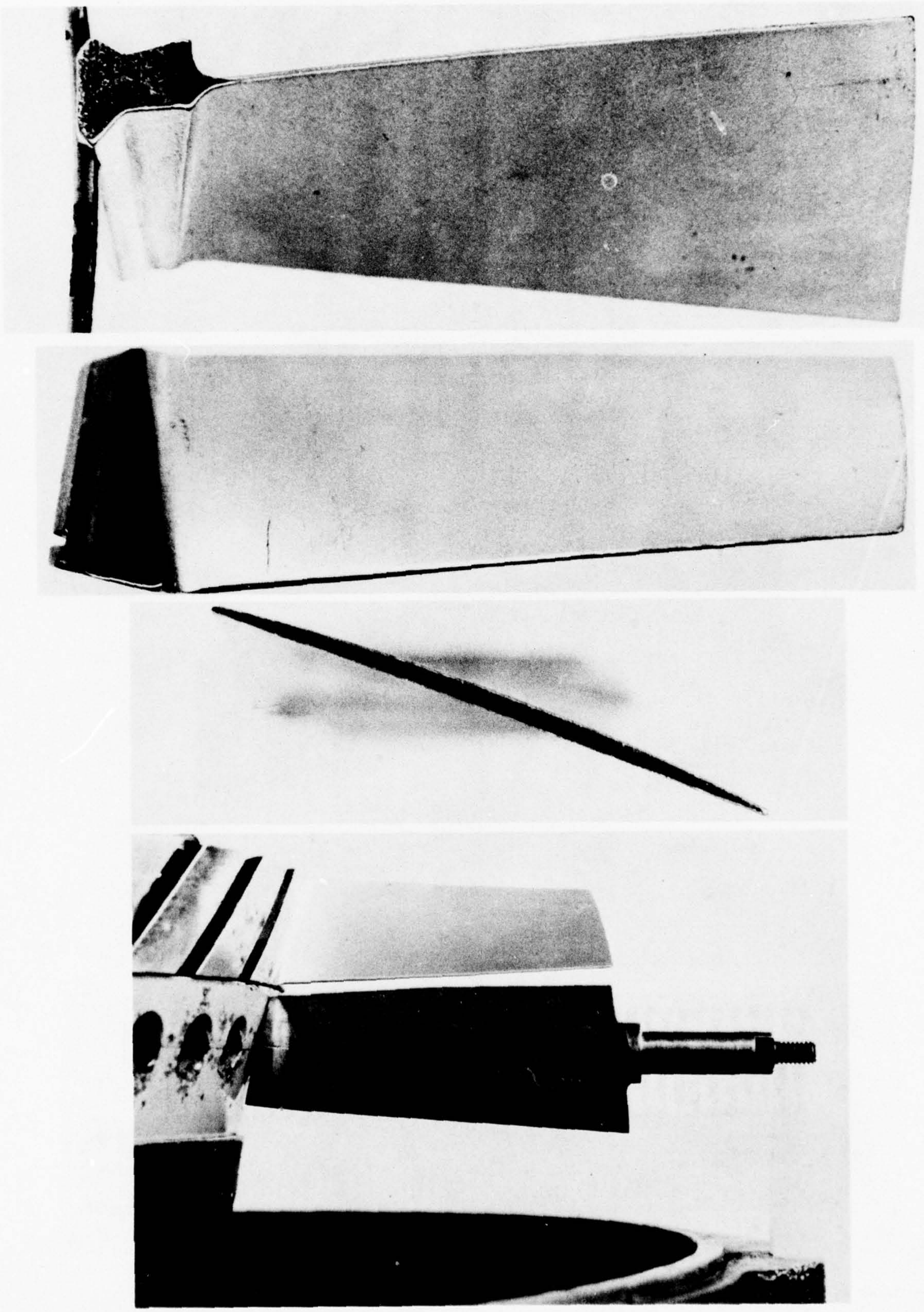

Table VI - First rotor blade and fixing ring (new and undamaged)

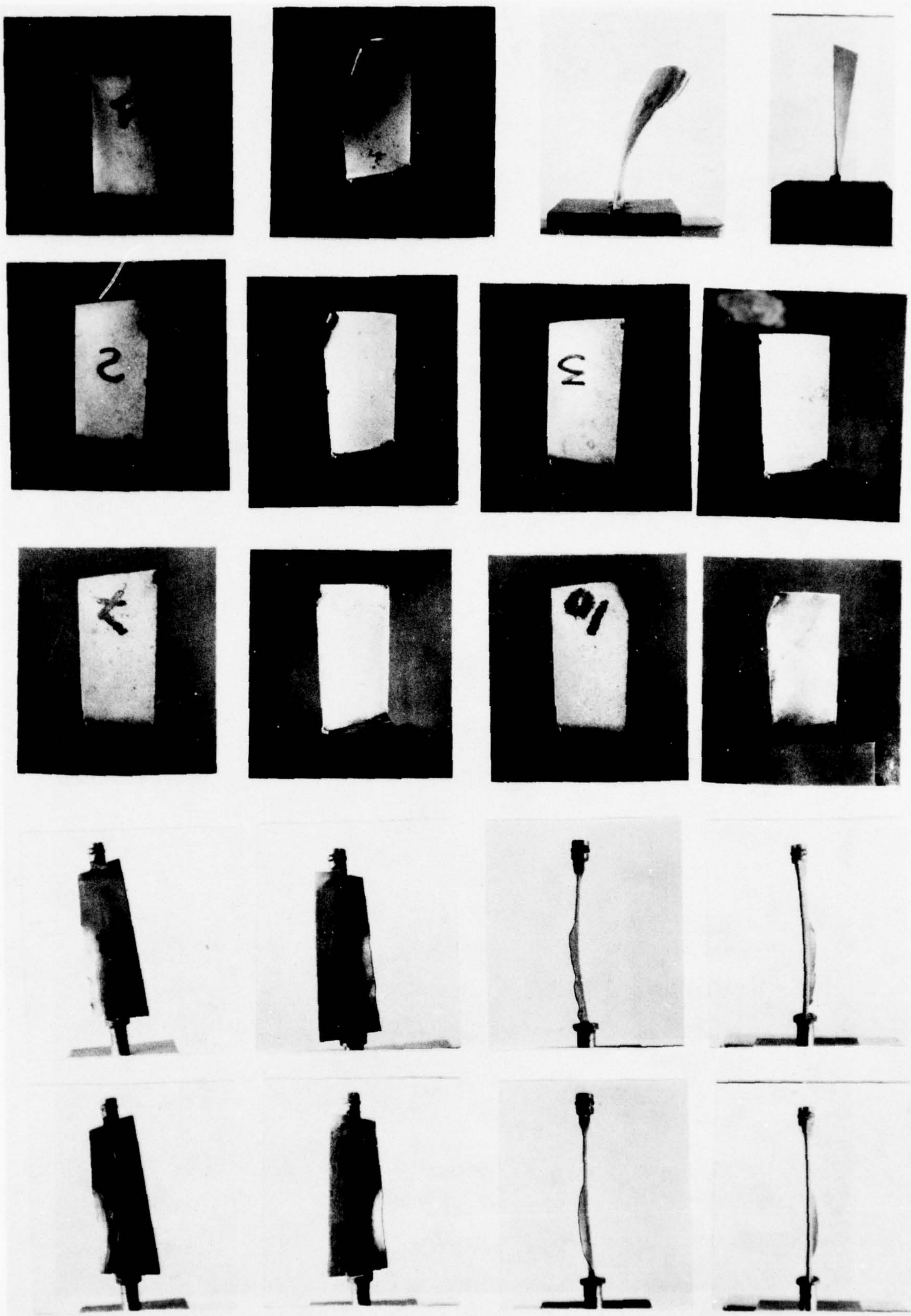

Table VII - Examples of first stage damaged blades and vanes

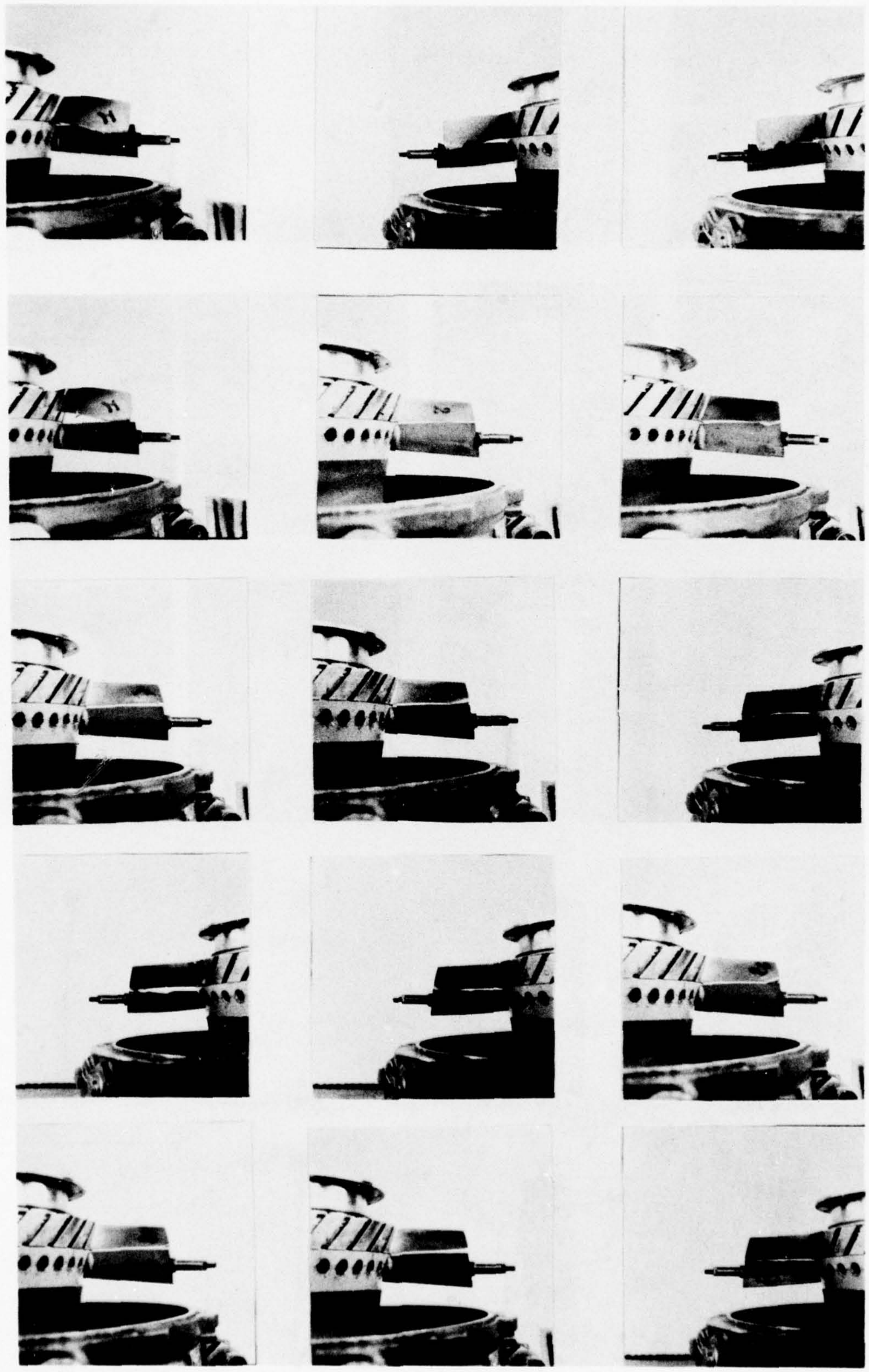

Table VIII - Mutual position of new and damaged vanes and blades of the first stage, with blade fixing ring released

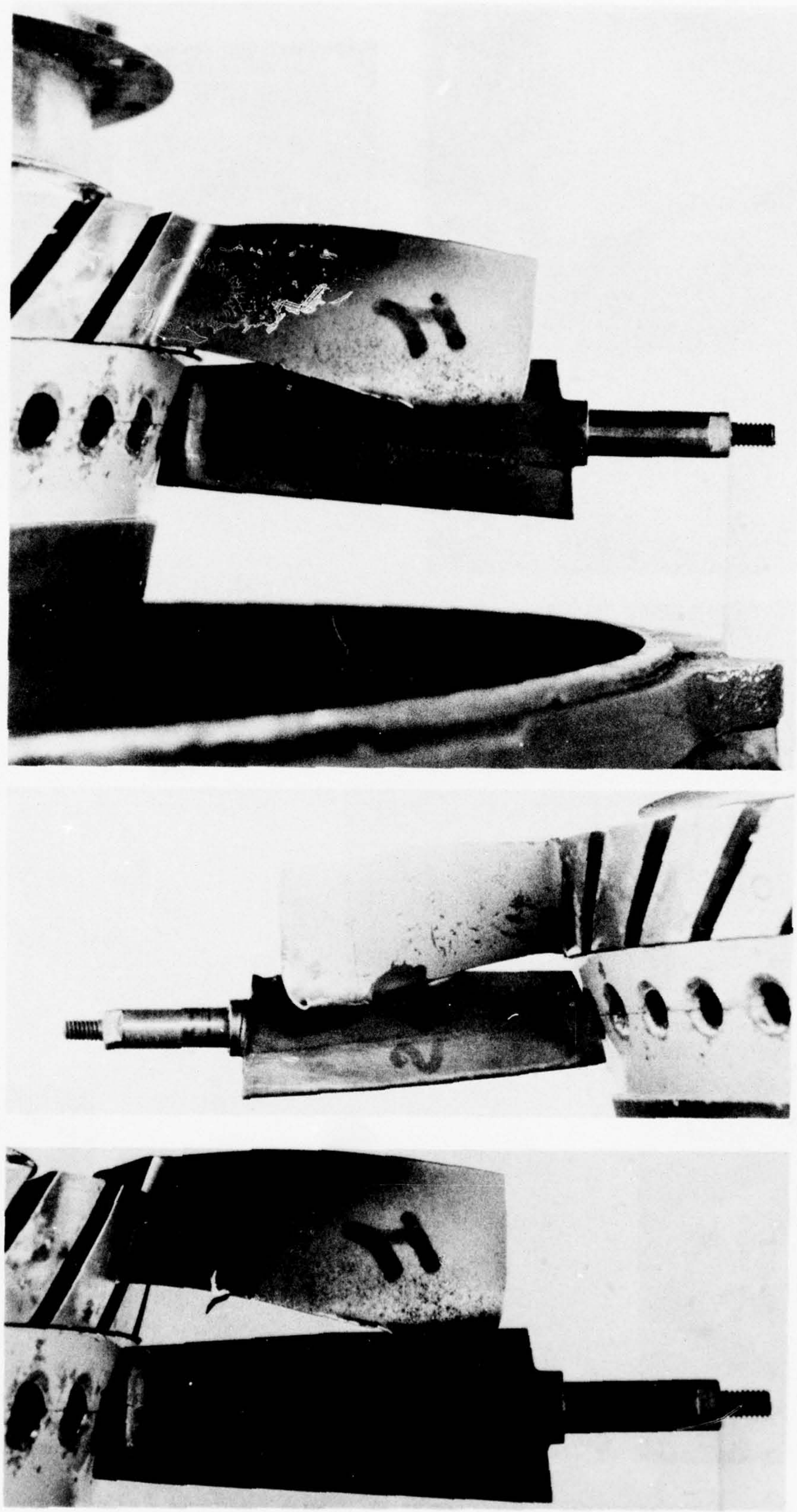

Table VIII bis - Mutual position of blade 1 and variable vanes

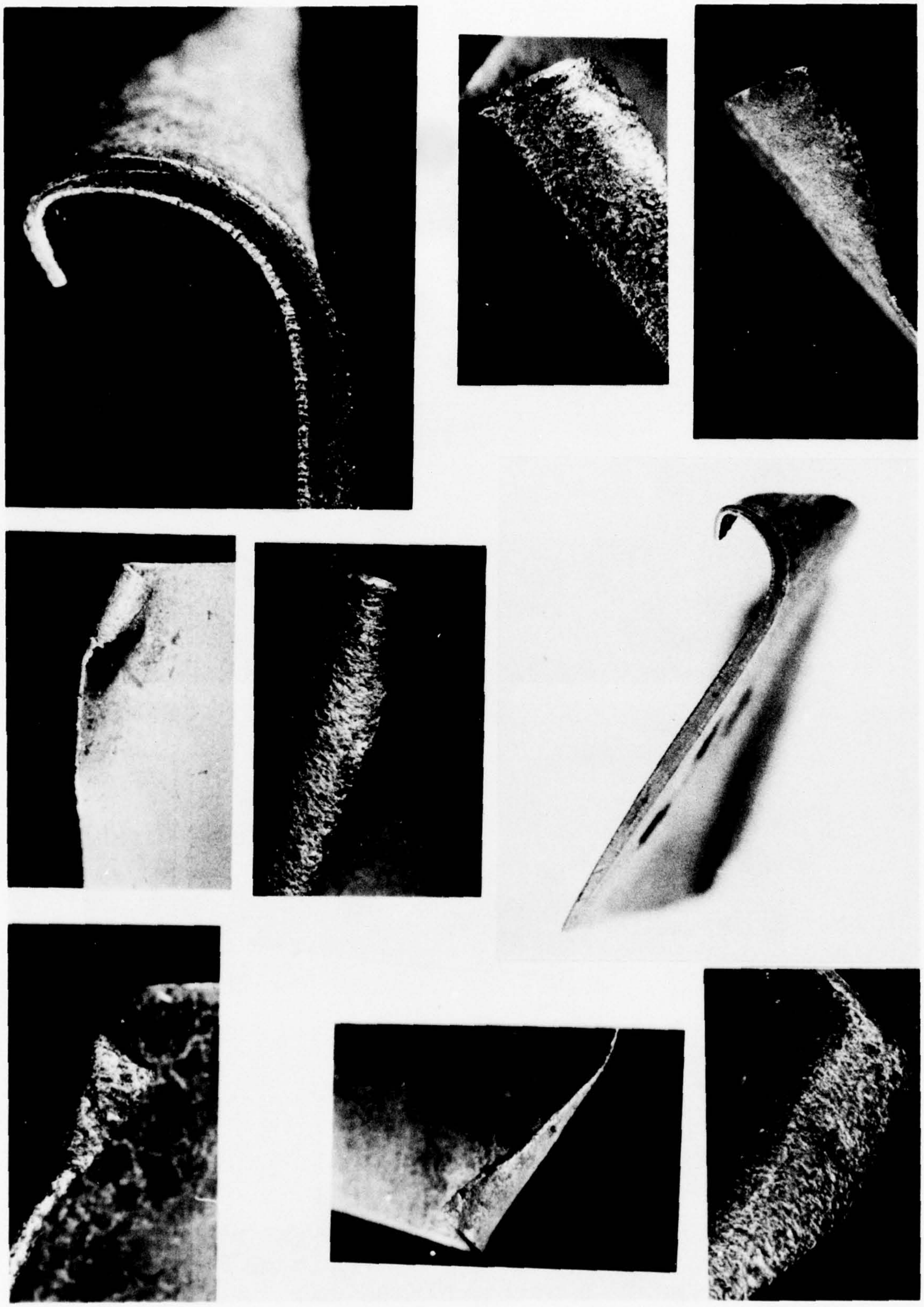

Table IX - Microphotographs of blade number 10, showing determined strains in three successive F.O.D. phases

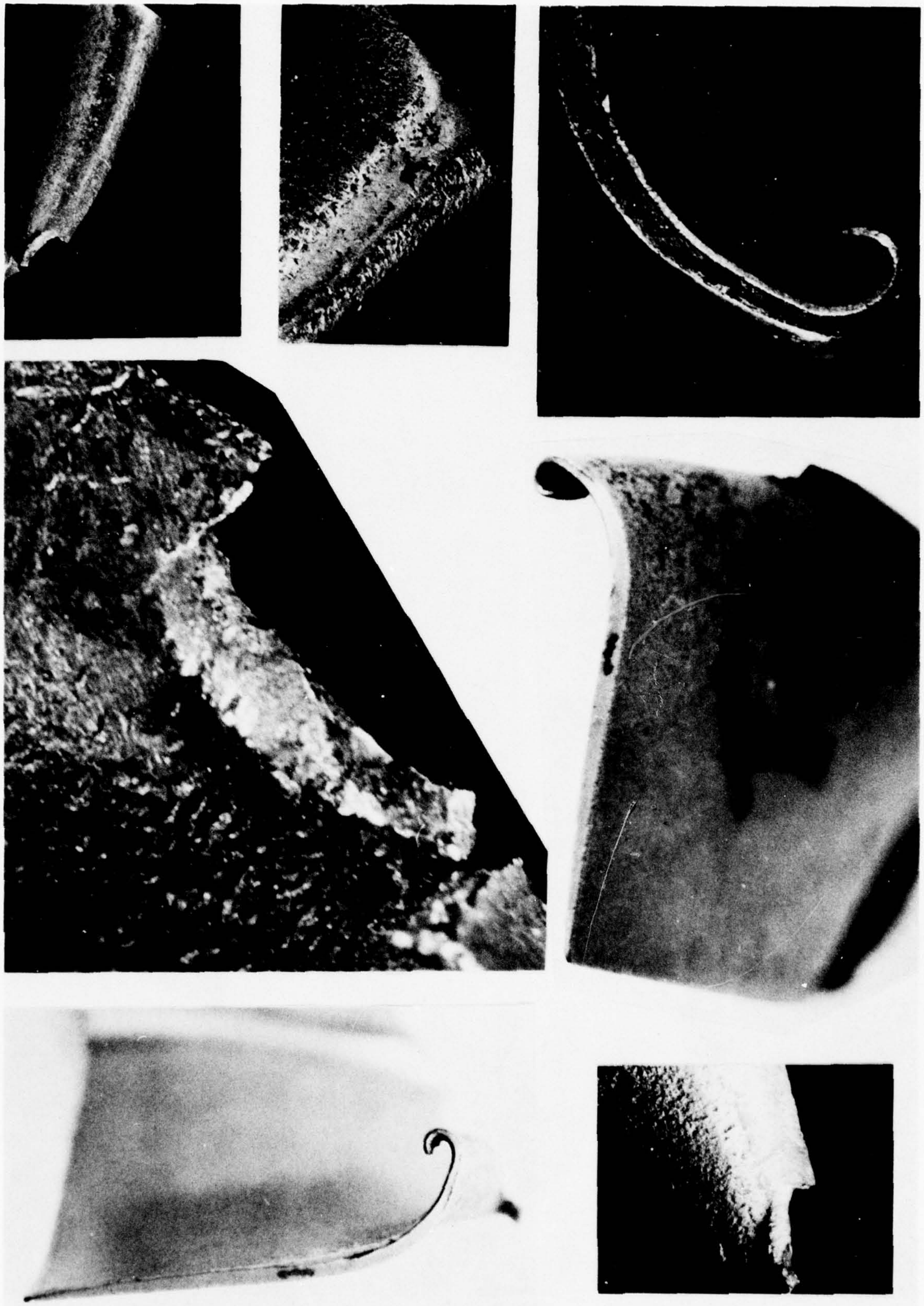

Table X - Macro and microphotographs of blade number 1, from which it was possible to rebuild the story of the previous and last strains

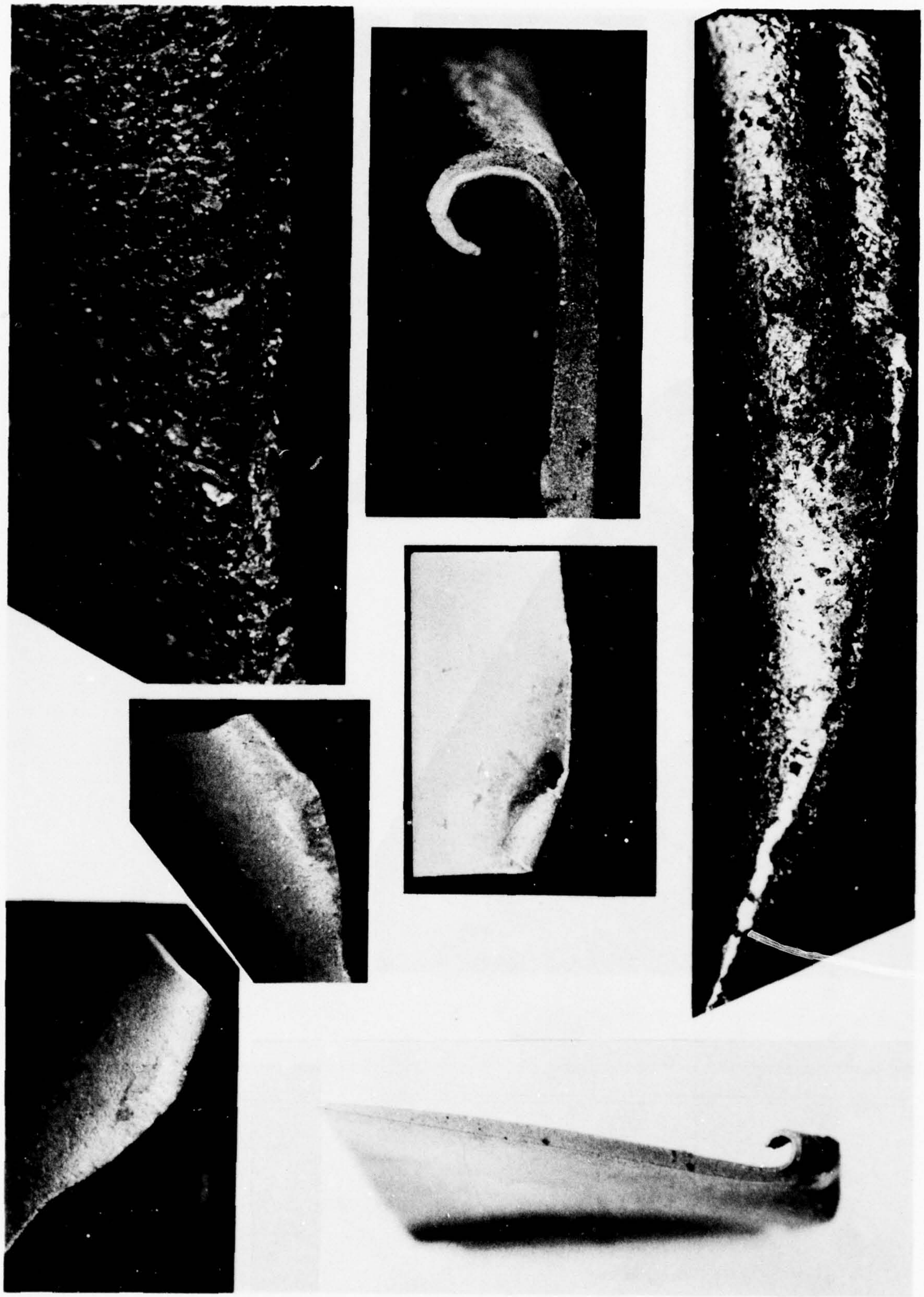

Table XI - Old and new strains in blade number 2

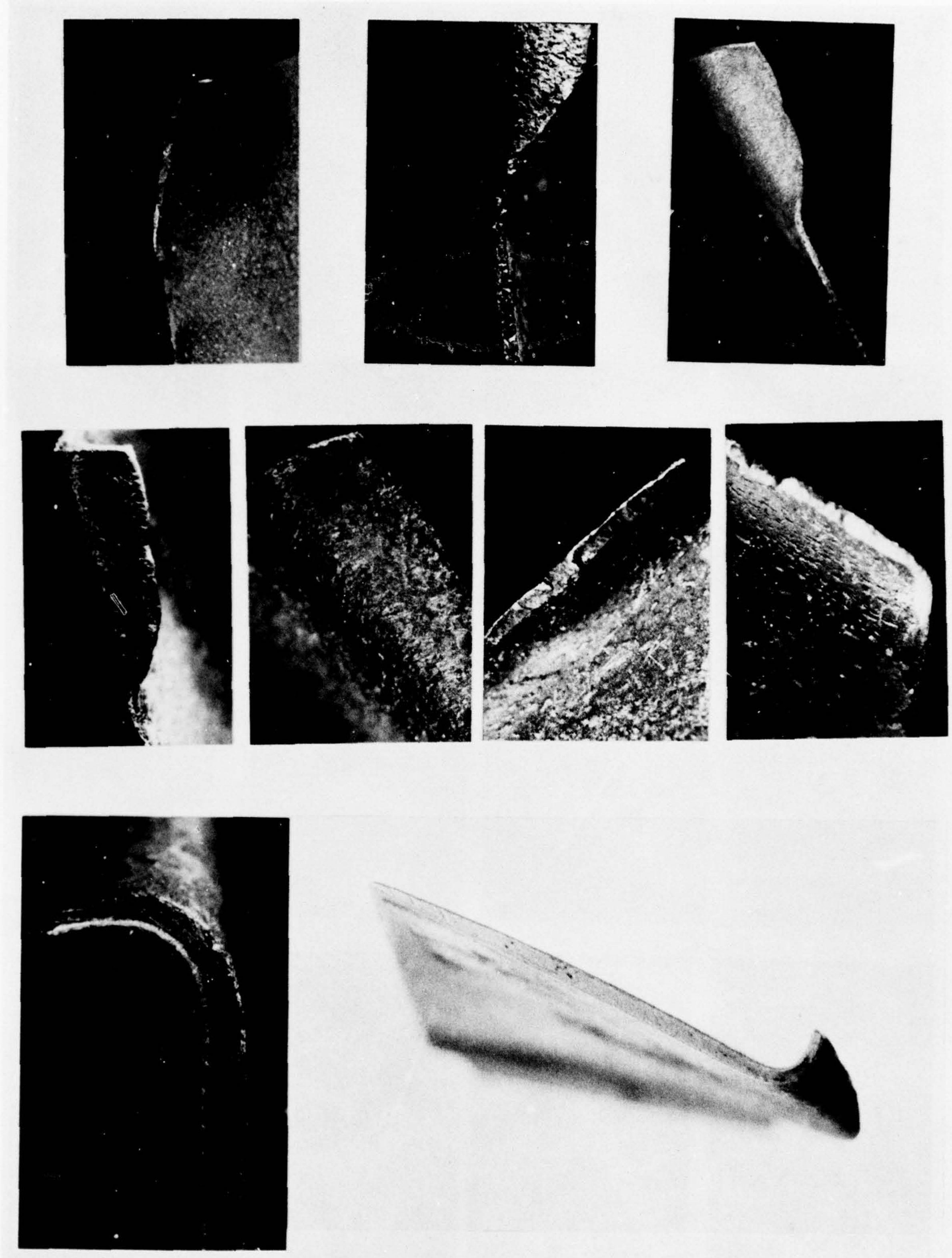

Table XII - Blade number 24, with yielding trace; blade number 25; blade number 26, reduced in size before the last running test

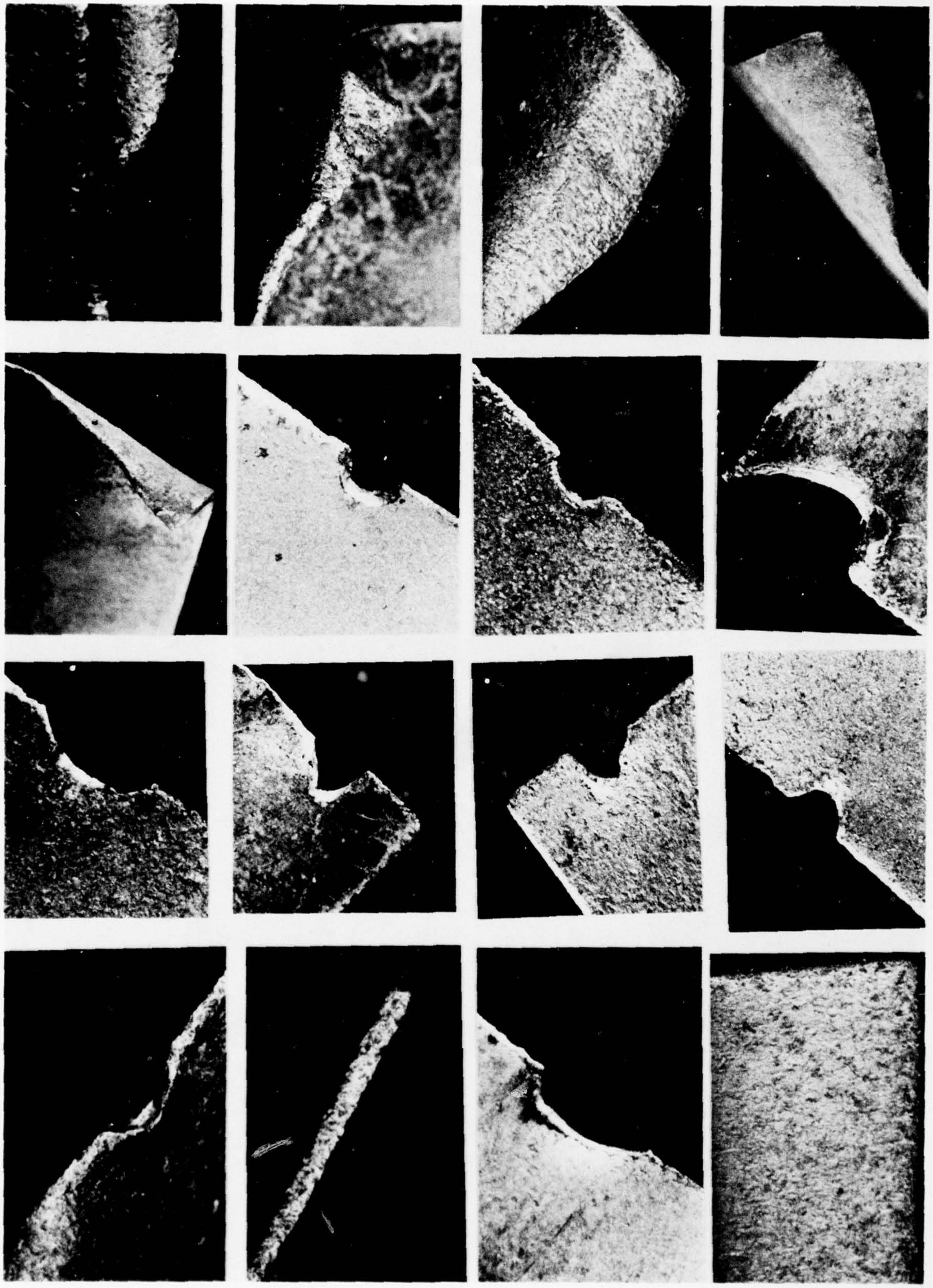

Table XIII - Typical simulated impact damages

PROGRES DE LA DETERMINATION DE LA VIE EN SERVICE PAR LES ESSAIS D'ENDURANCE

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RESUME

De nombreuses raisons poussent les constructeurs de moteurs d'avion à entreprendre des essais de longue endurance qui se distinguent notablement des essais habituels de développement et de certification.

La SNECMA a adapté la formule de l'essai cyclique accéléré à l'Ensemble d'Ejection qu'elle a conçu et mis au point pour le moteur Olympus 593 MK 610-14-28 de Concorde. Le lecteur trouvera une description du cycle d'essai comparé au cycle de vol, de l'installation au Centre d'Essai des Propulseurs à Saclay, France, et des précautions qui ont été prises pour assurer une reproduction aussi fidèle que possible des conditions de fonctionnement en utilisation.

Au moment de la mise en service commercial de Concorde, les cycles accumulés représentaient plusieurs années d'une exploitation qui devrait être sans surprises sur le plan de l'endurance.

OBJECTIFS D'UN ESSAI DE LONGUE ENDURANCE

Les spécifications techniques d'un type de moteur indiquent pour chaque élément ou sous-ensemble l'objectif de durée de vie totale à considérer au niveau de la conception et qui correspond à une bonne quinzaine d'années d'utilisation du moteur, soit de l'ordre de 40 000 heures pour un moteur d'avion de transport civil ou de 5 000 heures pour un moteur d'avion de combat.

Notre propos n'est pas de discuter la démonstration a priori de ces durées de vie qui implique essentiellement des calculs et des essais partiels.

Ce qui nous intéresse ici est la démonstration, par essai de moteur complet et avant la mise en service, d'une endurance couvrant les premières années d'utilisation, notamment en vue de :

- renforcer la confiance des futurs utilisateurs,
- vérifier la tenue du matériel en regard des garanties offertes,
- évaluer la dégradation des performances avec l'âge,
- apprécier l'endurance en fatigue des éléments qui ne peuvent être valablement sollicités en essais partiels (par exemple s'ils sont soumis à des gradients de température ou des spectres d'excitation acoustique complexes),
- mettre en évidence les usures ou détériorations que l'exploitant sera susceptible de rencontrer,
  - âge d'apparition (planning de disponibilité des pièces de rechange, ajustement du programme de maintenance recommandé, etc...)
  - mode d'apparition et d'évolution (méthodes de recherche des défauts et critères d'acceptation figurant dans le manuel de maintenance)
- valider les modifications ou les solutions de réparation préconisées,
- s'exercer à l'application des procédures d'entretien, d'inspection et d'intervention diverses,
- justifier la durée autorisée jusqu'à la première révision ou visite équivalente.

Cette énumération des objectifs assignés à un essai de longue endurance ne serait pas complète si l'on n'y ajoutait celui de maintenir, après la mise en service, le vieillissement réalisé chez le constructeur en avance sur le vieillissement réel en exploitation.

ESSAIS DE DEVELOPPEMENT ET DE CERTIFICATION

Les essais habituels de développement d'un nouveau type de moteur civil ne répondent pas de façon satisfaisante aux objectifs qui viennent d'être indiqués car,

- le volume total des heures d'essai allouées à l'ensemble du programme est en général relativement faible,
- une grande partie en est consacrée à l'évaluation des performances, des systèmes et du comportement mécanique, à la mise au point et à l'optimisation de la définition de type et non à des essais d'endurance proprement dits,
- la définition de type évolue notablement au cours du programme, de telle sorte que si les quelques premiers moteurs prototypes peuvent totaliser un millier d'heures, il est difficile d'utiliser cette expérience pour justifier l'endurance du standard de série.

De leur côté, les essais de certification visent essentiellement la démonstration de la sécurité dans des conditions extrêmes. Le point culminant du programme de certification est l'essai d'endurance de 150 heures. Cet essai, très traditionnel, a ses mérites mais n'apporte pas la solution car il est à la fois trop court, trop sévère et pas assez représentatif pour que ses résultats puissent être valablement extrapolés.

Le cas des moteurs militaires est quelque peu différent (nous pensons à ceux qui équipent des avions de combat). D'une part, pour aboutir à une durée de vie et une utilisation journalière égales environ au dixième de celles des moteurs civils, le programme de développement de ces moteurs est à peu près le même en volume. D'autre part, l'essai officiel d'endurance est également de 150 h, et ses résultats sont d'autant plus aisément extrapolables que son découpage est taillé à la mesure des missions envisagées.

CARACTERISTIQUES GENERALES DES ESSAIS DE LONGUE ENDURANCE

Donc, le constructeur d'un moteur civil doit définir un type d'essai de longue endurance, l'intégrer dans le programme de développement sans gonfler ce dernier exagérément, et lui affecter du matériel aussi proche que possible de la version de série.

L'objectif est de couvrir initialement de 2 000 à 5 000 heures de service, ainsi que le nombre de vols correspondants. Il s'agira par conséquent d'un essai cyclique accéléré qui négligera partiellement ou complètement les phases de vol qui ne consomment guère de potentiel horaire et n'imposent pas de variations de poussée qui affecteraient le potentiel cyclique.

On s'attachera à en faire un essai représentatif, ce qui implique que la succession et la durée des phases de vol retenues soient telles qu'elles reproduisent les gradients de contrainte et de température, sinon pour toutes les pièces, au moins pour les plus sollicitées. Cela suppose, entre parenthèses, que l'on ait acquis une bonne connaissance analytique et expérimentale de la réponse de ces pièces aux sollicitations stabilisées et transitoires. Le même souci de représentativité conduit à choisir des niveaux de charge du moteur tels qu'ils seront réellement sélectionnés en service et non pas des niveaux plus élevés, ce qui permettrait, certes, d'accélérer encore l'essai mais risquerait de fausser l'extrapolation par exemple en faisant apparaître des défauts non réalistes.

Nous allons décrire l'essai que la SNECMA a, dans cet esprit, fait subir à l'Ensemble d'Ejection du moteur Olympus 593 MK 610-14-28, le propulseur de Concorde, alors que Rolls-Royce, visant les mêmes objectifs, conduisait des essais cycliques spécialement adaptés au moteur de base. Durant tout le programme des essais de développement, l'Ensemble d'Ejection et le moteur de base n'avaient pas été dissociés, mais pour cet exercice en effet, il a été jugé préférable de les séparer pour les soumettre chacun aux conditions de fonctionnement qui le font le plus souffrir, sans allonger les délais.

DESCRIPTION DE L'ENSEMBLE D'EJECTION 14-28 DU MOTEUR OLYMPUS 593 MK 610-14-28

Les systèmes et sous-ensembles de l'Ensemble d'Ejection illustré par les Figures 1 et 2 réalisent les fonctions ci-après :

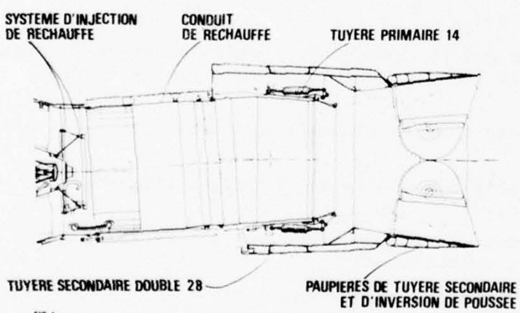

FIG 1

OLYMPUS 593 MK 610-14-28 ENSEMBLE D'EJECTION

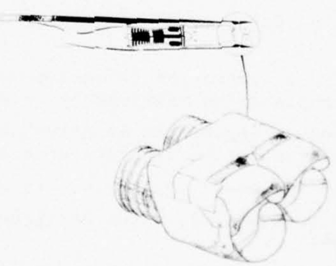

FIG 2

OLYMPUS 593 MK 610-14-28 ENSEMBLE D'EJECTION

Système ou sous-ensemble	Nombre par moteur	Fonction
Réchauffe	1	. augmentation de poussée programmée en fonction du régime moteur et de la température ambiante.
Conduit de réchauffe	1	. conduit de gaz entre moteur et tuyère primaire . siège de la combustion de réchauffe
Tuyère primaire à section variable	1	. Paramètre de régulation du moteur avec et sans réchauffe.

Système ou sous-ensemble	Nombre par moteur	Fonction
Tuyère secondaire, structure fixe	½ (une par paire de moteurs)	. continuité externe et interne des nacelles . col de reprise de l'écoulement primaire et de l'écoulement externe à chaque moteur . suspension des tuyères primaires (2) . articulation des 4 paupières . siège de l'inversion de jet
Paupières	2	. réglage de la section de tuyère divergente de chaque moteur en vol supersonique . réalisation de l'inversion de poussée par interposition dans le jet direct de chaque moteur.

L'une des particularités essentielles de la tuyère secondaire et des paupières est leur construction presque exclusive en STRESSKIN (panneaux formés à structure alvéolaire soudée par points, réalisés par la société américaine ASTECH), ce qui leur confère une grande rigidité pour une masse minimale.

La structure fixe de la tuyère secondaire est considérée comme une structure avion et, à ce titre, fait l'objet d'un essai de fatigue classique dans les installations de la British Aircraft Corporation, où lui sont appliquées cycliquement les températures et les charges mécaniques correspondant aux diverses phases de vol.

Cependant, la SNECMA a considéré qu'il fallait en plus, et malgré la complexité d'une telle entreprise, inclure cette tuyère secondaire double dans l'essai d'endurance dont nous allons parler, pour la soumettre effectivement à toutes les sollicitations engendrées par le fonctionnement des moteurs. A cela plusieurs raisons :

- le matériau utilisé ne bénéficiait que d'assez peu d'expérience dans des conditions sévères,
- l'essai structural ci-dessus mentionné ne reproduit pas la fatigue acoustique, facteur déterminant de la tenue des panneaux alvéolaires et ne recrée pas correctement le champ de température qui existe à l'arrière de la tuyère en inversion de poussée,
- il n'était pas envisageable de recourir à des essais partiels car chaque panneau représente un cas particulier fort complexe avec ses dimensions, formes, discontinuités locales, ses conditions aux limites et ses sollicitations thermiques et acoustiques propres.

CYCLE DE VOL

La figure 3 indique de façon schématique l'évolution typique du régime du moteur et du nombre de Mach en fonction du temps.

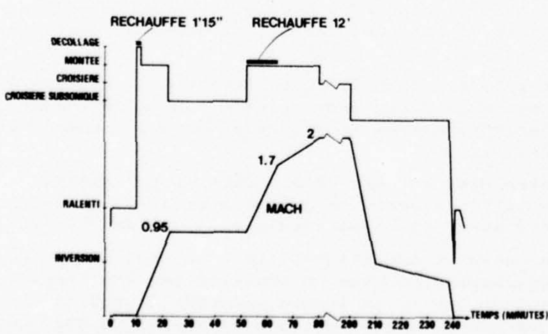

FIG. 3 - OLYMPUS 593 MK 610-14-28... CYCLE DE VOL

La réchauffe est utilisée :

- au décollage pendant 75 secondes en moyenne (durée maximale certifiée : 5 minutes)
- en montée, pour l'accélération transsonique, pendant 10 à 12 minutes (durée maximale certifiée : 15 minutes)
- au régime maximal d'urgence, au décollage, en cas de panne d'un moteur (durée maximale certifiée : 2½ minutes). Non représenté sur la figure 3.

L'inversion de poussée est utilisée :

- en vol à la manière d'aérofreins, pendant une durée maximale autorisée de 4 minutes, moteur au ralenti vol. Cette phase n'est pas représentée sur la figure 3.
- à l'atterrissage, pendant 5 à 15 secondes au régime maximal d'inversion (taux de détente : 2,05), puis au ralenti pendant environ 5 secondes. La durée maximale certifiée au régime maximal d'inversion est de 30 secondes.
- en cas d'accélération arrêt (décollage refusé) par sélection à partir du régime maximal de décollage (taux de détente initial : 2,35). Non représenté sur la figure 3.

CYCLE D'ESSAI DE LONGUE ENDURANCE DE L'ENSEMBLE D'EJECTION

Le cycle choisi par la SNECMA pour représenter un vol est illustré par la figure 4. Il comporte dans leur intégralité les trois phases qui sollicitent l'Ensemble d'Ejection de façon significative, c'est-à-dire le fonctionnement en réchauffe et en inversion de poussée à l'atterrissage, et se décompose ainsi :

Régime	Durée (minutes)	Remarques
Décollage	1½	Réchauffe préselectionnée au ralenti allumage automatique à la mise en puissance.
Croisière subsonique	2	Durée déterminée pour établir les conditions thermiques -peu élevées- dans cette phase.
Montée et accélération transsonique	17	Réchauffe pendant 15 minutes
Inversion maximale	¼	Taux de détente 2,3
Inversion ralenti	¼	
Ralenti	4	Durée déterminée pour rétablir les conditions thermiques au ralenti en poussée directe avant le début du cycle suivant.

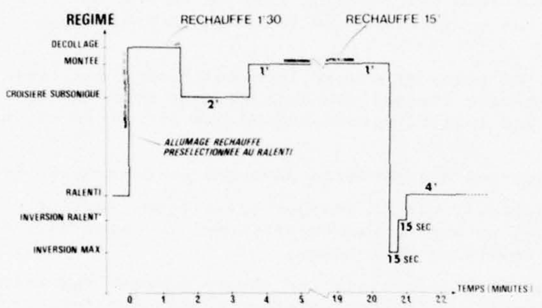

FIG 4 _ OLYMPUS 593 MK 610-14-28
CYCLE D'ENDURANCE DE L'ENSEMBLE D'EJECTION

La croisière, subsonique ou supersonique, non plus que l'inversion de poussée en vol, n'ont été représentées à cause des niveaux de pression et de température relativement faibles qu'elles impliquent. Le cycle ne comprend pas non plus de fonctionnement au régime maximal d'urgence de décollage ni au régime d'inversion de poussée en accélération-arrêt car il s'agit de cas rares qui ont par ailleurs été amplement démontrés par les essais de certification. Enfin, on n'a pas reproduit le cycle de mouvement des paupières en vol supersonique, en l'absence d'une simulation correcte des efforts aérodynamiques qui s'exercent dans cette phase, et qui, d'après les analyses effectuées, ne fatiguent pas la tuyère secondaire.

Le lecteur aura remarqué que sur deux points l'essai est plus sévère que l'exploitation courante :

a) la durée au décollage avec réchauffe est de 90 secondes contre 75. Comme il faut environ 70 secondes pour que les températures du conduit de réchauffe se stabilisent à leur valeur maximale, nous avons en réserve un coefficient de sévérité sur la durée cumulée supérieur au rapport 90/75.

b) le taux de détente en inversion de poussée est de 2,3 contre 2,05.

INSTALLATION D'ESSAI

L'essai a été réalisé au Centre d'Essai des Propulseurs (CEPr) à Saclay, France. Il était centré sur l'Ensemble d'Ejection et non sur le moteur de base qui servait en somme de générateur de gaz.

Bien entendu, installer la tuyère secondaire double et la solliciter correctement n'a pas été un mince problème. Elle est normalement chargée, par chacun des deux moteurs dont elle reçoit l'échappement, aux points de suspension des 2 tuyères primaires, d'articulation des 4 paupières et d'ancrage des 8 vérins de commande des paupières.

On a procédé d'abord à un essai d'évaluation avec deux moteurs côte à côte, fonctionnant simultanément -y compris en inversion de poussée- ou séparément, ce qui ne s'était sans doute jamais fait dans le monde auparavant dans un banc d'essai (poussée totale : 34 000 daN/76 500 lb).

Pour l'endurance proprement dite, il n'était nécessaire que d'immobiliser un seul moteur pour tester la baie correspondante complètement équipée. Cependant nous ne voulions pas que l'absence d'un moteur provoque des efforts anormaux (du point de vue de la longue endurance) dans la structure de la tuyère secondaire. Aussi avons nous conçu un système de mise en charge destiné à simuler dans la baie inoccupée les efforts venant de sa tuyère primaire, de ses paupières et de ses vérins de commande. Ces différents efforts étaient régulés cycliquement en fonction de la pression d'éjection du moteur de l'autre baie. Ce système a fonctionné à la perfection pendant tout l'essai d'endurance.

Concernant la baie directement affectée par le moteur, la comparaison des mesures de température et de vibrations -ces dernières d'origine acoustique et très intenses surtout en inversion de poussée- effectuées au banc et en vol en de nombreux points de la tuyère secondaire a permis de conclure à une bonne représentativité de l'installation.

Il n'a pas été jugé utile de préchauffer l'admission du moteur pour simuler l'échauffement dynamique car l'effet sur l'Ensemble d'Ejection est pratiquement négligeable, ni de reproduire le niveau de pression à l'entrée du moteur en montée et accélération transsonique car il est

inférieur à la pression atmosphérique au sol (d'où une intensité accrue du niveau sonore pendant les 15 minutes de cette phase en réchauffe).

RESULTATS

L'objectif de 1 750 cycles a été atteint au début de décembre 1975, soit presque deux mois avant l'entrée en service. Compte tenu des opérations de réglage s'ajoutant aux cycles d'endurance, nous avons réalisé :

3 566	allumages de la réchauffe
492 h 48'	en réchauffe
1 850	sélections de l'inversion de poussée
17 h 46'	en inversion de poussée.

Pour les éléments sensibles surtout au fonctionnement en réchauffe, nous considérons avoir simulé au moins 1 750 vols commerciaux, soit environ 5 000 heures sur la base de la durée de vol moyenne constatée aujourd'hui. Pour les éléments sensibles à l'inversion de poussée comme les parties arrière de la tuyère secondaire et les paupières, il s'est avéré après une étude statistique de l'utilisation commerciale que l'essai était plus sévère que la réalité (taux de détente 2,3 au lieu de 2,05). Les calculs, basés sur les mesures de température et de contrainte mentionnées plus haut et sur les caractéristiques en fatigue aléatoire du matériau STRESSKIN, ont montré que l'essai correspondait à 2 500 vols commerciaux, soit à peu près 7 500 heures.

L'essai s'est déroulé sans incident majeur, en deux tranches séparées par une inspection de détail programmée à 1 000 cycles (contrôle sonore du matériau STRESSKIN "à la pièce de monnaie", contrôles par endoscopes, gammagraphie ou radiographie). Quelques types de défauts, nouveaux ou déjà connus, sont apparus que nous avons laissés se développer ou que nous avons corrigés, profitant également de cet essai pour évaluer l'endurance de pièces réparées. Citons par exemple des décohésions locales et des criques de la peau extérieure du STRESSKIN sur les faces latérales externes des paupières et sur le côté interne de la partie arrière des murs latéraux de la tuyère secondaire. La parade consiste à protéger les zones affectées par des "doubleurs" en tôle fixés sur le STRESSKIN. Les paupières ainsi modifiées ont été livrées aux compagnies aériennes pour la mise en service, tandis que les murs latéraux sont maintenant pourvus de protections thermiques sur les zones susceptibles d'être affectées.

Revenant au problème de la représentativité de l'essai, on peut se demander, après plus d'un an d'exploitation commerciale, si ses résultats n'ont pas été infirmés en service réel. Actuellement, la réponse est non pour la grande majorité des pièces, mais ce n'est pas étonnant vu le nombre encore modeste des heures de vol accumulées. Cependant, nous avons eu tout de même une mauvaise surprise au sujet des accroche-flamme de réchauffe qui présentent des criques à des âges nettement plus faibles que celui que l'essai d'endurance nous avait laissé envisager. Ce point est intéressant parce qu'il montre que les conditions globales les plus sévères ne sont pas toujours les plus sévères pour toutes les pièces. Il s'avère que sur certaines routes on rencontre des températures ambiantes élevées en altitude. Cela conduit, par le jeu de la régulation du moteur et de la réchauffe, à une réduction de la richesse de réchauffe et ainsi à une flamme qui s'accroche de très près à l'accroche-flamme et le détériore rapidement. Or, pour l'essai, nous avions au contraire choisi des débits de carburant de réchauffe, donc des richesses, plus élevés dans le but évident d'imposer les températures maximales au conduit de réchauffe et aux tuyères primaire et secondaire.

POURSUITE DE L'ESSAI

Dès mi-76, le même Ensemble d'Ejection reprenait son essai d'endurance, selon un cycle modifié pour être également significatif pour certains éléments du moteur Olympus (addition d'une phase de croisière supersonique avec préchauffe de l'admission). L'objectif actuel est d'ajouter 850 nouveaux cycles.

CONCLUSION

La longue endurance de l'Ensemble d'Ejection de l'Olympus 593, en particulier celle de la tuyère secondaire, n'était pas évidente a priori, tant sont agressives les conditions de pression, température et vibrations auxquelles il est soumis. L'essai qui vient d'être décrit a permis avant même le début de l'exploitation commerciale de tabler raisonnablement sur des durées de vie effectives dépassant 5 000 heures pour la plupart des éléments.

L'endurance se poursuit, de façon à maintenir notre avance sur les utilisateurs, jusqu'au point où probablement l'expérience plus généralisée acquise par ces derniers viendra prendre le relais.

Malgré son originalité, cet essai s'inscrit dans la ligne générale des essais de longue endurance des moteurs que nous avons située au début de cet exposé. L'importance grandissante qui leur est donnée dans les programmes de développement des nouveaux moteurs devrait être perçue par leurs futurs utilisateurs comme un gage supplémentaire de leur fiabilité.

DISCUSSION

J.C.Ripoll

Vous avez mentionné (dans le texte) un cas où des essais sévères n'ont pas soumis toutes les pièces aux conditions les plus sévères. Comment peut-on éviter de tomber dans ce "piège".

Réponse d'auteur

Je crois qu'il n'y a pas de possibilités de faire un essai qui soit très représentatif pour toutes les pièces, et que, par conséquent, il est inutile de vouloir réaliser l'impossible; tout au plus peut-on identifier les composants qui ne seront pas essayés d'une façon satisfaisante au cours d'un essai donné, et l'essentiel sera de le savoir pour éventuellement faire un autre type d'essai adapté.

P.H.Gourgeon

Pensez-vous poursuivre ces essais jusqu'à la couverture de la vie complète de cet ensemble, ou sinon à quel moment pensez-vous vous arrêter?

Réponse d'auteur

D'une façon générale, le contructeur a besoin du feed-back de l'utilisation réelle pour réévaluer la représentativité d'un tel essai d'endurance. Dans le cas particulier de l'ensemble d'éjection de l'olympus nous allons marquer une pause et attendre une accumulation plus grande de l'utilisation commerciale et comparer ses résultats aux résultats de l'essai d'endurance.

D'autre part, cet essai ne sera pas poursuivi jusqu'a couvrir la vie complète car il ne peut finalement pas se substituer à l'expérience rélle et car la dépense serait trop importante.

ACCELERATED MISSION TEST - A VITAL RELIABILITY TOOL

by
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SUMMARY

The accelerated mission test has been successfully used in the F100 engine program to anticipate potential future problems. Early identification of service oriented problems has provided the lead time necessary to take corrective action before the problems occur in operation which decreases engine "down" time thereby improving life cycle cost.

The AMT is a supplemental testing procedure and must be used in conjunction with all of the advanced structural analysis techniques.

Plans are now being developed to conduct accelerated mission tests on engines that have completed the overhaul or depot cycle. The purpose of the testing will be to identify potential problem areas associated with engine parts that have been repaired in accordance with the overhaul procedures.

ABSTRACT

The powerplants of today's most advanced air superiority fighters must have a high thrust-to-weight ratio, rapid throttle response, low fuel consumption, low maintenance, functional efficiency over a wide altitude and Mach number envelope, and most importantly be reliable. To attain the desired reliability level requires new and innovative reliability techniques. This paper describes the fundamental constituents of the Accelerated Mission Test (AMT) instituted by Pratt & Whitney Aircraft in conjunction with the U. S. Air Force as a new reliability assessment tool on the F100-PW-100 augmented turbofan engine, the newest production engine in the U. S. Air Force inventory.

Accelerated endurance testing of aircraft engines is not a new concept. The innovation of the AMT is the accurate characterization of the engine duty cycle in the aircraft mission profile by the use of an Engine History Recorder (EHR), pilot interviews and analytically selecting the crucial events. Creating a cycle of only the engine damaging conditions (hot time, rotor speed, throttle cycles) in proportion to the mission can significantly increase the ratio of equivalent mission time to test time. This accelerated testing provides faster results at significantly lower costs.

Discussion

Engine performance, thrust-to-weight, is a critical element in the overall success of a fighter aircraft. Generally, the higher the engine performance is, the more effective the weapons systems. Hand in hand with performance goes reliability, for without this essential ingredient there can be no confidence in the total system.

Reliability doesn't just happen, it must be designed into the engine. Many assumptions must be made concerning the exact environment in which the engine parts must function and the higher the performance required, the smaller the margin for error during the design phase. Therefore, to realize the highest level of reliability in the design, accurate verification tests must be devised to assure the basic assumptions are correct.

The mission or duty cycle is one of the many assumptions that are made early in the design concept phase. The duty cycle is a prime contributor to the reliability of the propulsion system throughout its life. Historically, it has been very difficult to predict a real-life mission profile for a high performance fighter aircraft. The variable utilization of military aircraft (air-air, air-ground, etc.), is one reason for the difficulty in accurately predicting the mission profile. Another reason is that a wide variety of adversary aircraft frequently necessitate changes in tactics and therefore mission profiles on a continuing basis. The Accelerated Mission Test (AMT) program was devised for the F100 engine as a means of improving reliability throughout its service life. This program is composed of four parts: 1) determination of the actual mission profile; 2) defining test conditions; 3) AMT verification; and 4) implementing required engineering changes.

Determination of the Mission Cycle

The mission profile established for the F100 engine design competition was an almost equal mixture of subsonic and supersonic missions. The large amount of supersonic flight time underscored the need to concentrate on stress rupture life of the parts in the high temperature gas path. The subsonic throttle excursions of the engine were predicted on usage experience with high performance fighter aircraft in the early seventies. The F100 test requirements were predicated on these mission assumptions. The prototype F100 engine completed a 60-hour Preliminary Flight Rating Test (PFRT) in February 1972 that cleared the engine for the start of the F-15 flight test program. A rigorous 150-hour

Qualification Test (QT) was successfully completed in April 1974 by an operationally configured engine. The first operational F-15 was delivered to the Tactical Air Command in November 1974. Shortly after the F-15 became operational, a team of Air Force and Pratt & Whitney Aircraft engineers began a series of interviews with TAC pilots from the F-15 Tactical Training Squadron and the Operational Test and Evaluation organizations to identify the flight envelope, throttle movements, and time at various power conditions for the wide variety of missions conducted by the two organizations. The pilots readily admitted it was difficult to accurately recount the actions taken during the very active air combat missions. The F100 engine, however, has an electromechanical device, called an Events History Recorder (EHR) mounted on each engine. This device receives a variety of engine parameters that permit the clocking of total engine operating time, time at high turbine temperature and the number of throttle excursions. The information recorded by the EHR was then compared to the pilot interview information. Follow-up discussions were held with the pilots to verify that the engineers' data reasonably portrayed the mission being flown.

Figure 1 graphically shows how the test cycle is established. Periodic interviews are conducted to keep the engineering mission profile current with operational usage. For example, a change in mission mix ratio between air combat mission and ground support can significantly change the throttle excursion rates. This change is shown in Figure 2.

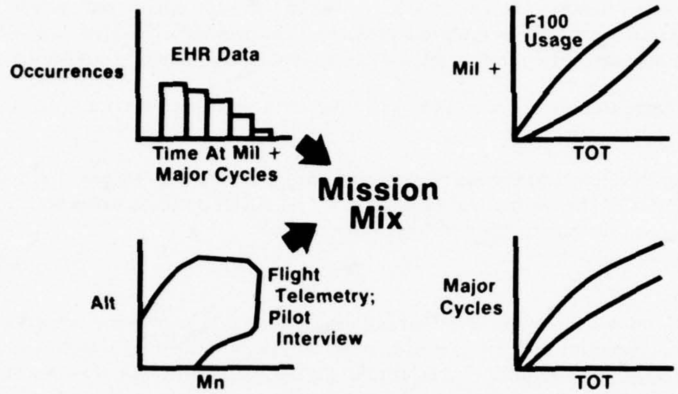

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Figure 1. Determine Actual Field Usage

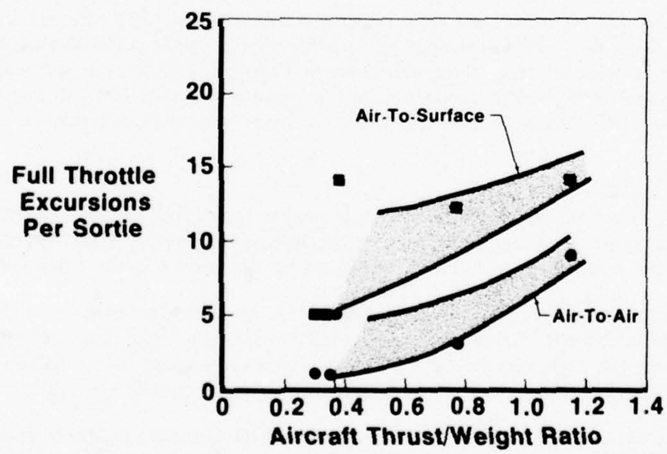

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Figure 2. Full Throttle Excursions Increase for Higher Thrust/Weight Aircraft

Defining Test Conditions

Analytical modeling based on stress and temperature surveys established structural life capability of all major engine parts with the new mission profile. The analysis indicated that operation at idle and cruise power conditions had an insignificant impact on engine life. As a result, idle and cruise power time was eliminated from AMT schedule; however, all full throttle excursions, high temperature time, and cold start take-offs were programmed into the cycle because their impact was significant. This technique permitted the compression of the test hours to mission hours by a factor of three and the savings in time and money is very significant. The AMT cycle is shown on Figure 3. During the test, the engine is subjected to supersonic time (altitude facility) proportional to the operational mission. Aircraft installation effects, horsepower and bleed extraction, and elevated fuel temperature are also programmed into the test. Cold start take-off cycles subject the hot section of the engine to the most severe environment. Figure 4 compared the

qualification test cycle to the present mission cycle. A hot parts temperature survey was conducted prior to the first AMT to establish the proper accelerated test technique. A full life cold start take-off cycle test (3500) was then completed before starting the AMT.

Obviously the test time compression factor varies with engine application and the design margins established for the engine. The compression factor for a bomber or transport engine would be significantly less and the throttle excursions would play a less prominent role in engine life usage ratio providing equivalent care was taken to maintain good transient turbine temperature control.

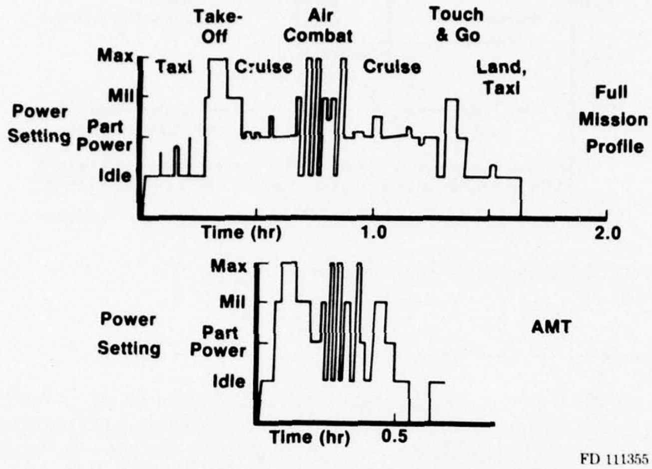

Figure 3. AMT Compression Saves Time and Money

	Spec or QT	AMT
Total time (hr)	2,000	2,000 (equiv)
Military and above (hr)	348	240
$\geq$ Mn 1.6 (hr)	70	2
Cold engine take offs	1,750	1,180
Partial cycles	350	10,700
Augmentor transients	987	7,600

FD 94805

Figure 4. Engine Usage is Significantly Different

AMT Verification

Air Force operational units have utilized a "Lead the Force" (LTF) program for newly deployed systems for many years. The Air Force has twelve F100 engines in an LTF program called "Pacer Century." The condition of the parts in the first AMT engine was compared to the parts in the first two Pacer Century engines at 250 hours and again at 500 hours. The AMT parts compared favorably to the Pacer engines at both inspection periods. This first AMT engine, which had some parts manufactured with development (soft) tooling, completed the equivalent of over four years of operational life (1300 hours) in 350 hours of actual testing in a little more than four months elapsed time. The test revealed the onset of thermal cyclic distress in the hot section at the 1000-hour inspection. The early problem identification has provided time for corrective action as shown on Figure 5. Two engines, taken directly from the production line, have been added to the program. AMT II has now completed 1750 hours of an air-air mission test and AMT III has completed inspection after 500 hours of an air-ground mission test. Hot section engineering changes designed to improve the distress found in AMT I are being evaluated in both of these tests.

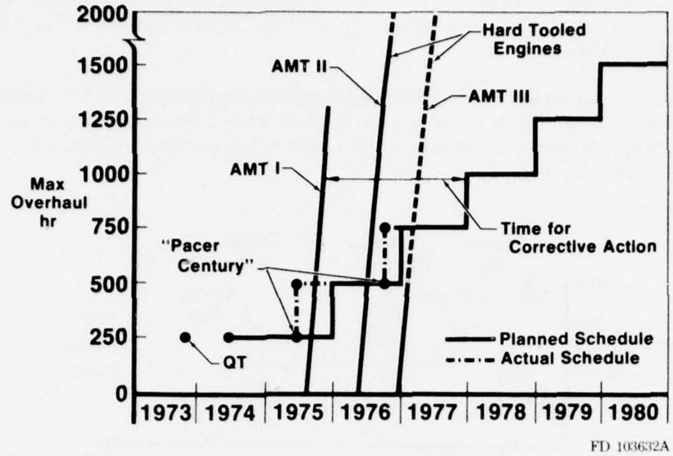

Figure 5. AMT Supports Pacer Century

Rapid Implementation of Changes

The primary purpose of the AMT is to improve the reliability of the engine during its full life time by identifying potential problems, developing improvements and implementing the changes during the early production phase of the program. The AMT has provided an insight into the future, thereby allowing time to take corrective action before the problems are encountered in service. Later, accelerated mission test engines will be used to prove out the engineering changes necessary to solve identified problem areas.

Benefits and Limitations

At the completion of the Accelerated Mission Test, a significant portion of the engine will have been subjected to conditions equal to that expected during its service life time. The AMT does not, however, expose all of the engine to a full life time of usage because it is an *accelerated* test. The gearbox, bearings and seals, for example, will be subjected to far more revolutions during their service life. The AMT is not a replacement of the normal design analysis and laboratory verification tests such as bearing and seal rigs. Further, the AMT program is a very small statistical sampling of the operational engines. The AMT by itself is limited in scope, however, combined with advanced structural analysis techniques such as finite element technology and design sensitivity analyses, it becomes a very strong reliability assessment tool not previously used.

DISCUSSION

J.Slatford

The author mentioned that the AMT did not cover very adequately the HCF type vibration. May I ask if Pratt & Whitney have any ideas on how this type of vibrations can be covered by engine testing?

Author's Reply

Yes, one method that is used is small step changes in speed and watch for vibration effects. The actual qualification test, or endurance test, is structured in that manner. We intend to use that type of testing to determine whether we have a critical vibratory mode.

E.E.Abell

If I may comment on checking of the HCF mode:

The high cycle fatigue modes are evaluated by means of strain gage instrumentation on special test engines. These engines will be operated over the entire flight envelope. The results from these tests are examined thoroughly to insure that no deleterious vibration modes exist in the gas path airfoils. Such programs have been accomplished on current development USAF engines.

M.Holmes

What proportion of accelerated mission test is carried out in altitude test facilities?

Author's Reply

The equivalent (to the testing) to the data that we receive from the field. We conduct test from Mach Number 1.6 to Mach Number 2.0. Now in that particular regime the Mach Number excursions in an altitude test stand are not totally representative of what you see in operation. The altitude facility cannot make the transitions as rapidly as the aircraft can in operation. The amount of time is, as I showed in my chart. The time above 1.6 is only two hours in two thousand hours.

J.A.Aguer

What is the order of magnitude of turbine and compressor disc lives on military engines as compared to certified disc lives on commercial engines like JT8D's and JT9D's?

Author's Reply

I have made a comparison of the F100 for 2000 hours of operation compared to JT8D. And that comparison compares very favourably from LCF type cycles to 8000 hours of operation in airlines. It is amazing how close it is.

W.Overmars

Is aircraft intake distortion represented during accelerated mission testing?

Author's Reply

No, we do not run distorted tests. We do not feel that that is representative from a structural stand point. We do use hot fuel and horse power extraction to make it representative to what happens in the field however. We do apply horsepower extraction and bleed extraction that the airplane uses and also hot fuel. This engine has to operate with 200°F fuel inlet temperature.

W.Overmars

Is total duration AMT's as indicated on your charts the equivalent of numbers of flight hours shown or have actual test hours been shown?

Author's Reply

It is 2000 hours of equivalent aircraft operation not actual run hours.

W.Overmars

You have indicated that your AMT cycle simulating 1-1/2 hours flight in 1/2 hour test. So is your actual AMT running 2000 hours therefore simulating 6000 cycles?

Author's Reply

My chart shows the total number of (start) take-off cycles, and the number ends up to be 1180 flights.

G.N.J.Robinson

Reverting to the question of high cycle fatigue testing, we have found that one of the principal problems is to identify the precise location of the maximum stresses. I wonder if you have considered or you are accustomed to use brittle lacquers as a means of determining the regions of stress concentration or any other technique?

Author's Reply

I think we are using strain gauges on engines as Mr. Abell said, we do a complete stress survey of the engine at the initial phase of its program, and then if there are changes we go back and do stress surveying of the engine as well as thermal surveying. But we do not use stress lacquer.

M.Holmes

The missions assumed in the engine specification and the qualification test schedule were found to vary from actual engine usage; has this been anticipated in the design stage? How much overdesign resulted?

Author's Reply

As I mentioned earlier we designed the engine to the QT requirement but we also had a mission applied. As a matter of fact, it is very difficult at times to get a good representative mission at the beginning of a program. And so we made efforts earlier in the program phase to establish some form of cyclic life that we thought we had to have. So we did design the engine to a greater HCF cyclic life than the QT required. In the process of designing the engine we designed it to very severe stress rupture characteristics.

In addition, we had it designed for our theoretical cycle or mission that we had placed upon ourselves which was somewhat of the order of 4000 cycles compared with the 10000 that we now are seeing. We had to go back and do a complete reanalysis of the engine from a standpoint of thermal and mechanical stress for all discs and for all critical parts in the engine.

The last question is related to whether we had the engine overdesigned from the standpoint of weight. We have designed from a stress rupture standpoint and we have added cooling flow to the turbine in excess of what would be necessary, had we known the mission cycle. I do not feel though with the thrust to weight ratio that the engine has obtained, that we have compromised too far.

E.E.Covert

I think this question illustrates a significant difference between designing engines for military use and designing engines for commercial use. I think it is much easier to be able to say with some confidence what the used cycle will be for a commercial engine. The primary reason of course is obvious but the people who drive fighter type airplanes when they have a new tool become highly skilled in using this tool to its full advantage. And this use may not correspond to what the designers had in mind because the designer does not think like the young fighter pilot and of course the young fighter pilot does not think like the designer either.

EXPERIMENTAL INVESTIGATION ON THE INFLUENCE OF COMPONENT FAULTS ON TURBOJET ENGINE PERFORMANCE

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SUMMARY

In this paper some results of the experimental investigations on the effect of implanted local faults, e.g. turbine guide vane damage, plugged fuel nozzles and turbine rotor blade damage, on the performance of a single spool turbojet engine are presented. The formation of flow non-uniformities downstream the faults is especially described.

In one-dimensional gas path analysis systems circumferential averaged thermodynamic parameters are used for fault detection. The effect of the implanted faults on some of these averaged parameters will be shown in comparison with the local parameter changes in the disturbed sector.

Furthermore the possibilities of using the analysis of flow non-uniformities for the isolation of local faults in the hot section of turbojet engines are discussed and questions of probe position for this diagnostic technique are ventilated.

LIST OF SYMBOLS

F	thrust
$\dot{m}_F$	fuel flow
p_F	fuel injection pressure
p_{t2}	total compressor discharge pressure
p_{t4}	total turbine discharge pressure
p_4	static turbine discharge pressure
T_{t1}	total compressor inlet temperature
T_{t4}	total turbine discharge temperature
φ	circumferential angle
ρ_F	fuel density

Subscripts

c	corrected
O	without fault

1. INTRODUCTION

Today the "health" of a turbojet engine in an aircraft is estimated with the help of a few parameters, such as fuel consumption, exhaust gas temperature, engine speed and throttle lever position. In case that the parameters during the flight diverge from the nominal values, due to an engine fault, statements about the nature and of the extent of the fault in general can only be made after a trouble shooting on the ground, i.e. visual inspection, borescope check, oil analysis and so on.

If one wants to obtain during the flight detailled information about an engine fault in order to take the necessary steps to end the flight mission successfully, to avoid secondary damages and to give the required maintenance orders after landing, additional engine parameters have to be taken into consideration and to be analysed.

The monitoring of the mechanical "health" of an engine and of the accessories is possible to a great extent by means of the appropriate fixing of sensors. It is, however, very difficult to detect faults in the gas path components, such as compressor, combustion chamber and turbine. A lot of engine faults can be isolated by measuring and analysing thermodynamic parameters. These possibilities, however, are limited by the generally necessary limitation of the number of parameters, by the achievable measuring accuracy

and by the nature of diagnostic system. In order to detect failures which are for the most part homogeneous and which can be seen in a change of efficiency, of mass flow and in a change of area, diagnostic systems which are based on an one-dimensional method suffice.

2. LOCAL FAULTS CREATE NON-UNIFORMITIES

Especially faults in the hot section of an engine like inhomogeneous combustion caused by a malfunction of burners, faults in the mixing zone of secondary and primary air in the combustion chamber and turbine guide vane damages create intensive temperature and flow non-uniformities. The isolation of such local faults is hardly possible by means of an one-dimensional analysis. When using single probes completely contrary changes in parameters according to the position of probe may be the result, so that from the following diagnosis too contrary conclusions have to be expected. If the measured values of several probes are averaged, the changes in parameters due to such faults can become zero.

Flow non-uniformities can also be caused by disturbances outside of the engine like local compressor air bleed or compressor inlet flow distortion. Experimental investigations on the effect of circumferential temperature and pressure distortion at compressor inlet on engine performance showed especially marked non-uniformities of temperature at compressor exit and a change of compressor characteristic /2/. In this paper however, only the effects of faults within the engine are considered.

The inclusion of an analysis of temperature and pressure non-uniformities can be a valuable help for local fault isolation, as already mentioned by Fuhs /3/. The detection of such faults at their beginning is certainly a matter of great importance, because the danger of subsequent damages - e.g. caused by local overheating - is very great and may cause an engine shutdown.

3. EXPERIMENTAL INVESTIGATIONS AND RESULTS

In order to examine the effect of local faults on flow and performance of a turbine engine, defects were implanted in the area of combustion chamber and turbine. The investigations were carried out with the single spool turbojet engine ATAR 101 F2 on the institute's test bed. This ATAR engine has a 7-stage axial compressor, an annular combustion chamber with 20 duplex burners and a 1-stage turbine.

At compressor exit there are 14 total pressure/total temperature probes and static pressure holes. At turbine exit, just behind the rotor, 8 combination probes with 3 radial measuring points for total pressure and total temperature were installed (Fig. 1) and wall holes for measurement of static pressure were made. By revolving the turbine exit casing it was possible to measure the thermodynamic parameters of turbine exit flow at 24 different circumferential points. For test data recording and processing the test bed is working on-line with a computer.

The test data "with fault" were corrected and related to the data "without fault" with the same corrected engine speed and the same exhaust nozzle position, e.g.:

$$\Delta T_{t4c} = \frac{(T_{t4}/T_{t1}) - (T_{t4}/T_{t1})_0}{(T_{t4}/T_{t1})_0} \cdot 100\ \%$$

3.1 Plugged Fuel Nozzles

For the simulation of an unequal burning in the combustion chamber six fuel nozzles were supplied with magnetic valves, so that the fuel flow leading to these six nozzles could be interrupted.

Fig. 2 shows the effect of "plugged fuel nozzles" on the temperature distribution at turbine exit at 90% corrected engine speed and open exhaust nozzle. The changes in total turbine discharge temperature are plotted against the circumferential angle. The number of fuel nozzles which are shut off is 10%, 20% and 30% of total fuel nozzle number. Already if 10% of the burners are plugged up a marked non-uniformity in temperature can be observed.

Due to the considerable increase of fuel flow and together with the increase of fuel pressure in the operating burners - if some fuel nozzles are shut off - the temperature profil in the undisturbed sector of the combustion chamber changes too. So there are also some temperature non-uniformities in the undisturbed sector at turbine exit in case of great faults, e.g. 30% of fuel nozzles shut off.

When correlating the position of the faults at turbine inlet and the temperature or pressure non-uniformities at turbine outlet, one must keep in mind that there is a displacement of the flow in the turbine of about 20 degrees in the direction of rotation.

The changes in total pressure distribution at turbine exit caused by plugged fuel nozzles are plotted in Fig. 3. The asymmetry of the curves is typical of this kind of disturbance. This asymmetrie of pressure is caused by the interactions of the distorted ("cold") and the undistorted ("hot") flow in the turbine stage. As the velocity vector diagrams and the simplified streamlines in Fig. 4 qualitatively show, there is a decrease in absolute flow angle at turbine exit, due to the lower velocity level of the "cold" flow down-

stream of the fault. Therefore in the marginal zones of the distorted flow there is in direction of rotation a decrease at first and then an increase of throttling of the turbine. This effect is confirmed in Fig. 5 by the changes in static pressure distribution at turbine exit.

It is remarkable, that the nature of the noted non-uniformities of turbine discharge pressure and temperature are not dependent from the extent of the fault.

Non-uniformities of compressor discharge pressure and thus an unequal throttling of the compressor in circumferential direction could not be registered. By the compensating effect of the annular combustion chamber the fault does not influence the engine components upstream.

As a result of plugged fuel nozzles the fuel injection pressure increases. According to the laws of incompressible flow the fuel flow is proportional to the pressure differential at the fuel nozzle:

$$\dot{m}_F \sim \sqrt{(p_F - p_{t2}) \cdot \rho_F}$$

In Fig. 6 the difference of the injection pressure and compressor discharge total pressure is plotted against the fuel flow. One can see that also in case of great faults -30% of the fuel nozzles are shut off - there is a clear linear relation. The data "without fault" were taken during a longer test period considering the different densities of fuel.

Thus the fuel injection pressure coefficient $\sqrt{(p_F-p_{t2})\rho_F}/\dot{m}_F$ can be used for detecting such defects in the fuel system, which cause an increase or a decrease of flow resistance downstream of the fuel pressure measuring point. Furthermore leakages in the fuel system can be detected by monitoring this parameter.

Consequently an exact isolation of the fault "plugged fuel nozzles" is possible with the help of the analysis of temperature and pressure non-uniformities at turbine exit and the help of the fuel injection pressure coefficient.

3.2 Turbine Guide Vane Damage

For the investigation of the effect of a turbine guide vane damage on the flow conditions upstream and downstream of the fault, 3 of the 29 nozzle guide vanes were removed. Fig. 7 illustrates the extent of the implanted fault.

The measured effects of this fault on the thermodynamic parameters at turbine exit are demonstrated in Fig. 8 for one engine operating point. In the undisturbed sector an increase of the turbine discharge temperature by 8% can be registered although the total pressure in this area remains unchanged. Due to the lower flow deflection - lower work of expansion - in the disturbed sector of the turbine, there is a remarkable increase of the total temperature (34%) and the total pressure (19%).

In contrast to the "cold" flow by plugged fuel nozzles the distorted flow due to a turbine guide vane damage has a higher velocity level than the undistorted flow. With the same engine speed the flow angle in the disturbed sector at turbine exit becomes greater, so that the interactions of the distorted and the undistorted flow are contrary to those demonstrated in Fig. 4. Therefore the static pressure curve in Fig. 8 first rises and than falls in direction of rotation. In the zone at about 90 deg. circumferential angle, downstream of the first guide vane at the side of the "hole" (missing 3 vanes), the unguided flow along the suction side of this blade will be separated for the most part and cause high flow losses. Therefore the effects of flow with low velocity on the adjacent flow with high velocity in this zone are superposed on the interactions mentioned above. Thus at about 90 deg. in Fig. 8 the static pressure rises again after the marked decrease of pressure.

Moreover one can see, that due to the interactions of the distorted and the undistorted flow the circumferential extent of the flow non-uniformities is greater than that of the fault.

As in case of the plugged fuel nozzles, non-uniformities in the compressor discharge pressure also could not be observed in case of the turbine fault and there was no influence on the compressor operating point too.

The different colouring of the turbine stator between bucket ring and turbine shaft (Fig. 7) indicates that in the area of the fault hot gas was streaming between stator and rotor disc. This is illustrated in Fig. 9 and is due to the smaller spin of the distorted flow at stator exit. A decrease in spin correlates with a change in radial pressure distribution, so that there is an increase of static pressure at the hub.

When within multi-stage turbines the primary fault remains undetected, the mentioned local overtemperature at the exit of the frist turbine stage can lead to subsequent damages at the nozzle guide vanes of the following stage.

The measured values of total temperature and total pressure plotted in Fig. 2, 3 and 8 are average values of 3 radially distributed measuring points. Fig. 10 shows the total

temperature changes and Fig. 11 shows the total pressure changes at 77%, 50% and 23% of the blade height.From the qualitative viewpoint the curves at the various radiuses do not differ significantly. Only the quantitative parameter changes in the distorted sector are a little greater at 77% blade height.As for the fault "plugged fuel nozzles" the influence of the radial position of the probes on the noted parameter changes at turbine exit is negligible. So, for the detection of the characteristic non-uniformities caused by a fault the measuring of the circumferential distributions of turbine discharge total temperature and total pressure at one radius suffices.

3.3 Turbine Rotor Blade Damage

For simulating a turbine rotor blade damage the trailing edges of 5 blades and opposite to them of 6 blades were partly out off. Fig. 12 illustrates one part of the implanted fault.

The result of the investigations was, that due to this damage at the same corrected engine speed and the same exhaust nozzle position there is an increase of fuel consumption of about 2% and an increase of thrust of about 1,5%. The measured changes of temperature and pressure at compressor exit and at turbine exit were smaller than 1% and thus they were within the accuracy of measurement.

The influence of the turbine rotor fault on the acceleration behaviour of the engine was much greater. Fig. 13 shows the effect of the fault on the acceleration behaviour at part load. For a step change of throttle lever position with low overfuelling the increase of acceleration time is about 6 seconds. A further result: for a certain engine speed change the difference of acceleration time due to the fault becomes greater, when the overfuelling allowed by the fuel control unit becomes lower.

4. CONCLUSIONS

Fig. 14 gives a summary of the changes of parameters caused by the studied faults: turbine guide vane damage and plugged fuel nozzles.

As expected, one of the most important parameters for engine condition monitoring is the fuel consumption. The fuel injection pressure coefficient, which is independent of the engine operating point, gives very important information about the state of fuel system and fuel nozzles. Also the thrust is of great importance for diagnostics, but only if measured directly.

Fig. 14 also shows the changes of the parameters: total temperature, total pressure and static pressure at turbine exit and compressor discharge pressure, all circumferentially averaged. Furthermore it is illustrated that the local changes of these parameters in the area of flow non-uniformities created by faults are considerably greater compared with the averaged parameters. The example "plugged fuel nozzles" demonstrates very clearly, that an isolation of this fault is only possible with the help of the analysis of flow non-uniformities.

Summarizing one can say that the inclusion of the analysis of non-uniformities into engine diagnostics makes the isolation of local faults in the hot section of turbojet engines much more promising. When using this technique, total temperature and total pressure are of great importance for fault detection, whereas the static pressure is only of secondary importance.

Of course it should not be unmentioned, that for the isolation of little faults a great number of circumferential measuring points is necessary. Although there is a problem of probe quantity, problems of measurement are facilitated by the fact that nature and position of the non-uniformities are of primary significance, while the quantitative parameter changes are only of secondary significance.

For the detection of temperature non-uniformities for example, it suffices to measure the local temperature changes in relation to the circumferentially averaged temperature. Thus, in order to detect faults, the use of baseline data (data of the "health" engine) for the relation are not necessary.

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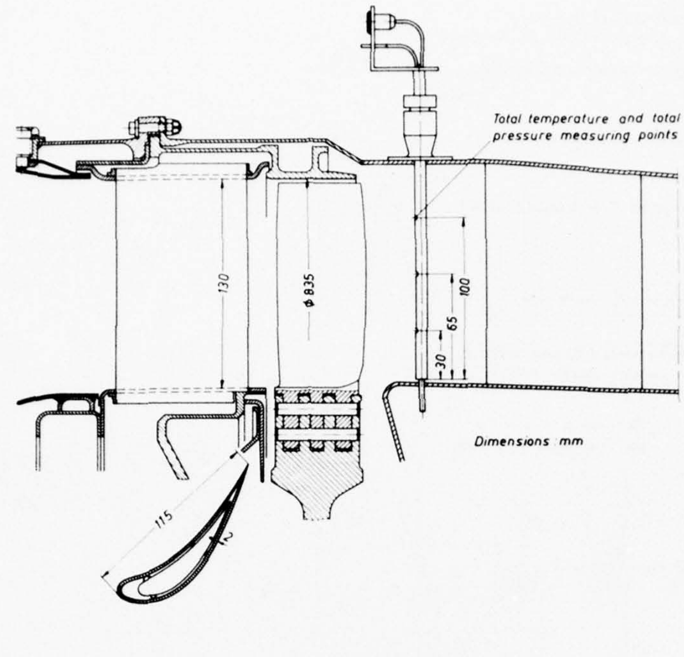

Fig. 1: Turbine stage of the ATAR engine, position of the combination probes

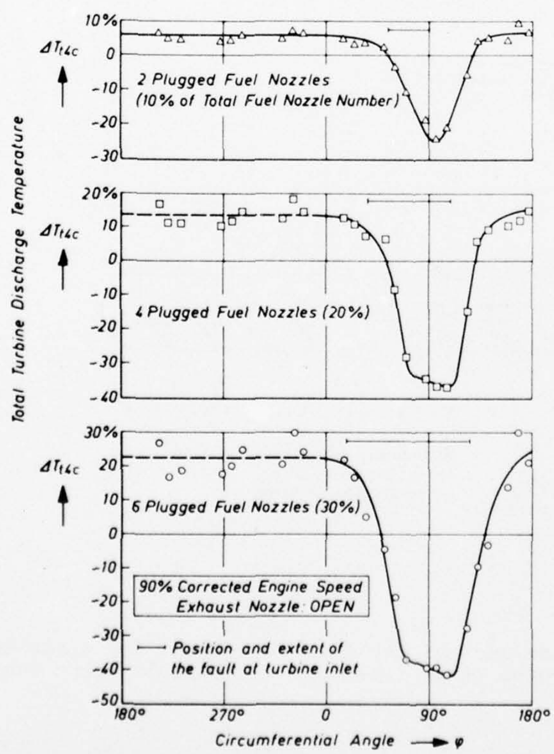

Fig. 2: Changes in total turbine discharge temperature, "Plugged Fuel Nozzles"

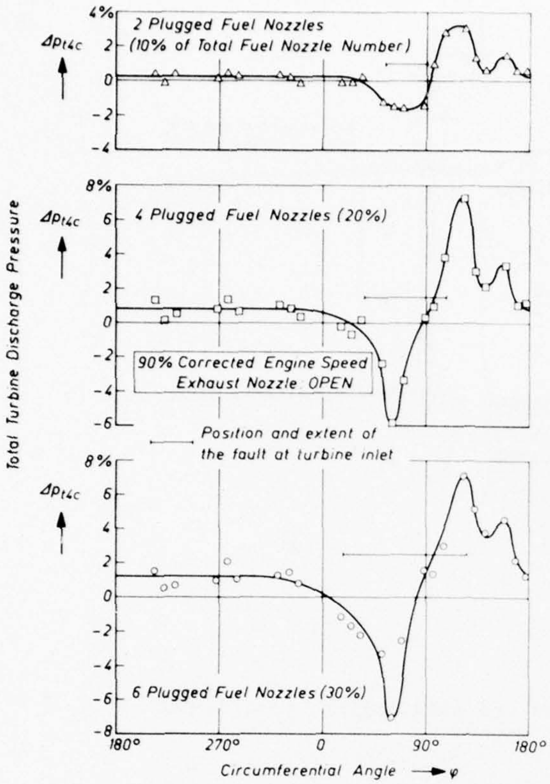

Fig. 3: Changes in total turbine discharge pressure, "Plugged Fuel Nozzles"

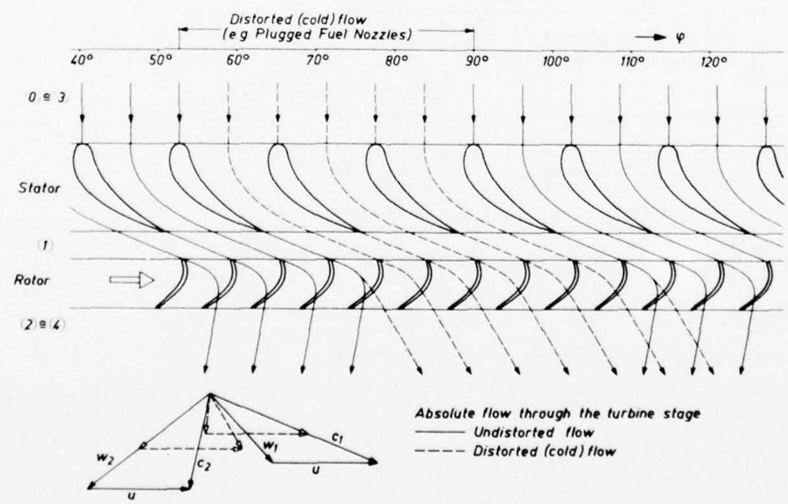

Fig. 4: Simplified streamlines of the undistorted and the distorted flow in the turbine stage (absolute system), "Plugged Fuel Nozzles"

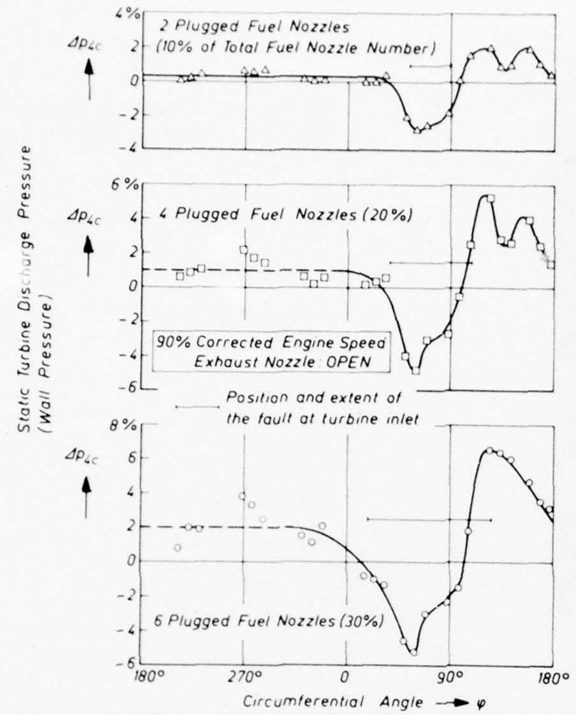

Fig. 5: Changes in static pressure at turbine exit casing, "Plugged Fuel Nozzles"

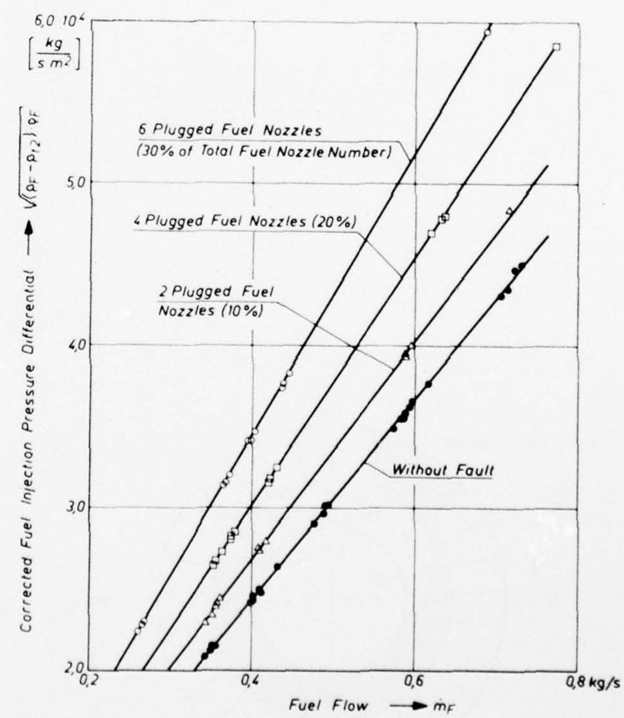

Fig. 6: Fuel injection pressure differential versus fuel flow, "Plugged Fuel Nozzles"

Fig. 7: Implanted "Turbine Guide Vane Damage", 3 vanes removed

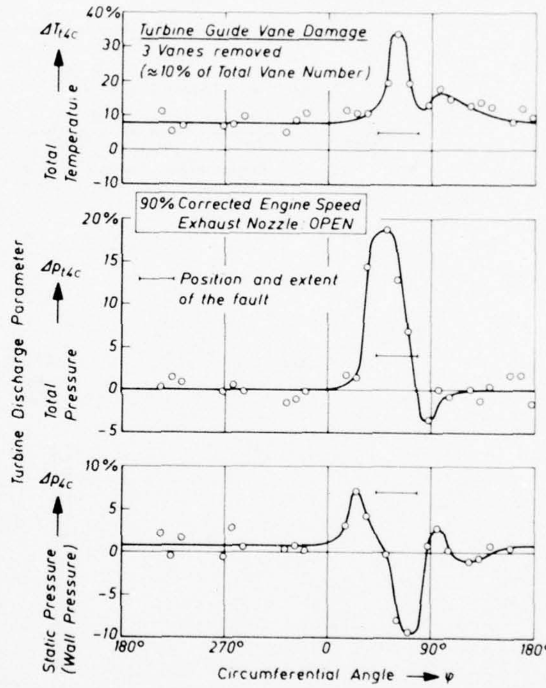

Fig. 8: Changes in total temperature, total pressure and wall pressure at turbine exit, "Turbine Guide Vane Damage"

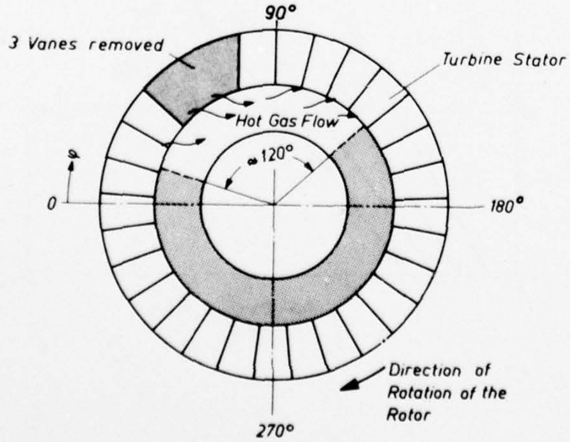

Fig. 9: Inward flow of hot gas between turbine stator and rotor disc due to "Turbine Guide Vane Damage"

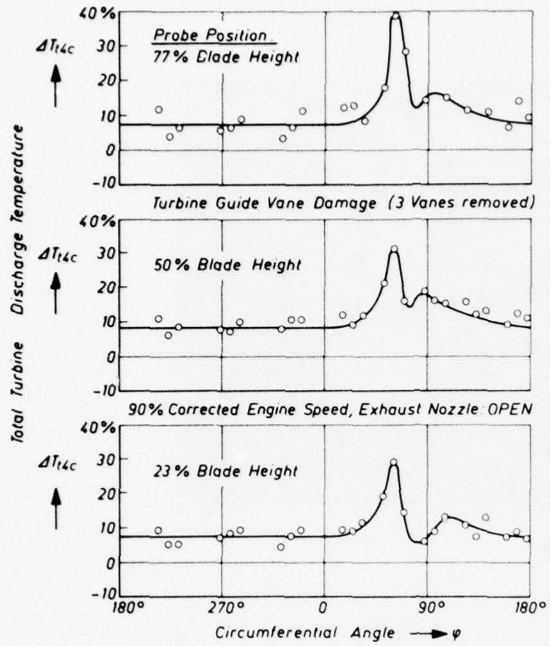

Fig. 10: Changes in total turbine discharge temperature at different blade heights, "Turbine Guide Vane Damage"

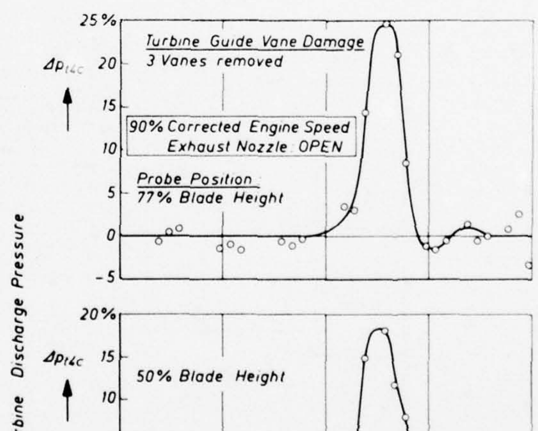

Fig. 11: Changes in total turbine discharge pressure at different blade heights, "Turbine Guide Vane Damage"

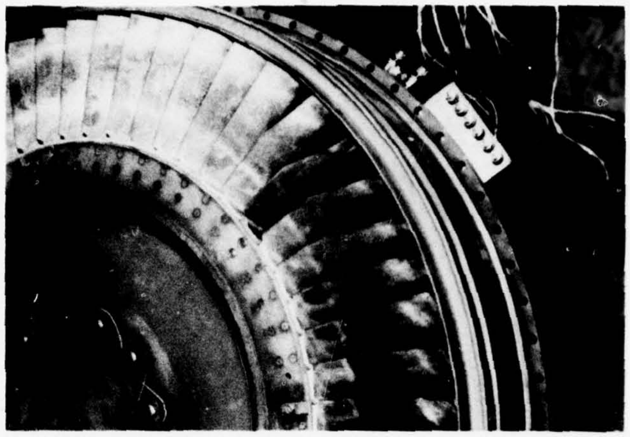

Fig. 12: Implanted "Turbine Rotor Blade Damage"

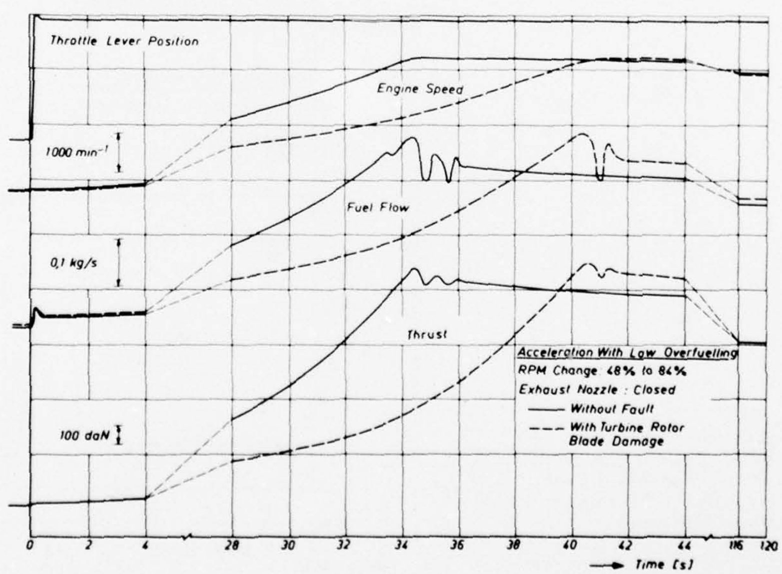

Fig. 13: Engine parameters response for a step change in throttle lever position with low overfuelling, "Turbine Rotor Blade Damage"

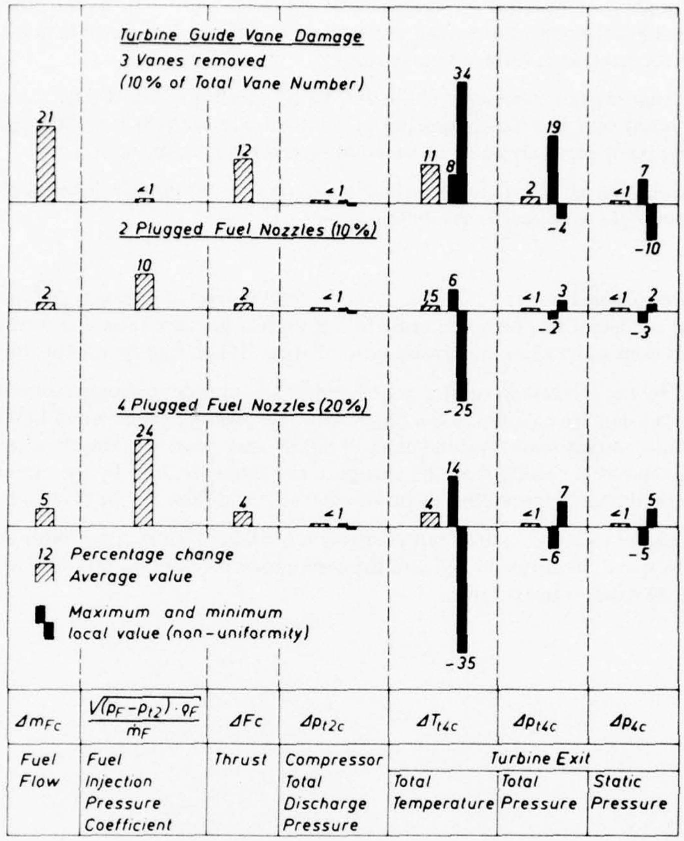

Fig. 14: Changes in averaged parameters and changes in local parameters due to local faults implanted in the hot section of a turbojet engine

DISCUSSION

H.I.H.Saravanamuttoo

I am somewhat surprised that removal of three out of 29 nozzle guide vanes did not show any difference in the compressor operating point. The effect of decreasing the blockage on the compressor would be expected to lead to a drop in compressor delivery pressure at a fixed compressor speed. It is possible that some compensation of this may come from an increase in turbine temperature.

It is noted however, that the tests were carried out at 90% compressor speed. Could the author please indicate the % thrust at this speed, and also the compressor pressure ratio at both 90 and 100% speed. The effects on compressor operating point are likely to be considerably greater at max speed.

It is agreed that circumferential variations of turbine temperature are a good indication of combustion distress, and this is used on some gas turbines in gas pipeline service.

Author's Reply

The investigations were carried out at 72%, 84%, 90%,and 96% corrected engine speed for different exhaust nozzle positions. At no operating point a decrease in compressor pressure ratio due to the fault could be observed. There was even an increase in pressure ratio of about 1% at high speed and closed exhaust nozzle.

This effect is caused by the increase in turbine inlet temperature in order to compensate the loss of turbine performance due to the damage and due to the drop in turbine pressure ratio caused by the higher temperature level at turbine exit at constant mass flow and exhaust nozzle area. Furthermore the change of the effective aerodynamic area is somewhat smaller than the change of the geometric area by the removal of 3 nozzle guide vanes. That is because the undistorted flow at turbine stator exit influences the flow in the disturbed sector.

Therefore at 100% engine speed not a drop but probably a small increase in compressor discharge pressure will be observed. At 90% speed the thrust is 70% and the compressor pressure ratio is 80% compared with 100% speed and for the same exhaust nozzle area.

METHODS OF IMPROVING THE PERFORMANCE RELIABILITY OF ADVANCED MILITARY POWER PLANT SYSTEMS

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SUMMARY

Advanced military propulsion systems will be equipped with multi-parameter control systems based around an electronic computer making use of digital technology. Such control systems have a high degree of flexibility and can be used for a wide range of applications. This offers new possibilities of efficient and accurate monitoring of engine performance in the aircraft and by the flight support organisation responsable for reliability and safety of the propulsion system. The possibilities will be discussed based on the experience gained with a multi-parameter electronic engine controller of a V/STOL fighter aircraft. Special consideration will be given to practical aspects such as handling procedures.

LIST OF SYMBOLS

CAS	calibrated airspeed	kts
CBV	control-bleed valve	
EGT	turbine exit temperature	°C
FF	fuel flow	lb/h
IGV	inlet guide vane	
M_∞	flight mach number	
N	force, thrust	Newton
N	lift engine speed	% RPM
N_1	speed of LP-shaft	% RPM
N_2	speed of HP-shaft	% RPM
p_∞	ambient pressure	mb
P	total pressure	
P_1	total pressure at compressor face	
P_3	total pressure at exit of HP compressor	
t_∞	ambient temperature	°C
T	total temperature	
T_i	total temperature at compressor face	
T_x	total temperature fan exit	
TAS	true airspeed	kts
V_∞	flight speed (indicated)	kts
η	$N_2 / (N_2$ at $N_1/\sqrt{T_x}$-limit$)$	
σ	swivell-nozzle deflection angle	°
φ	fuel flow / fuel flow at $N_1/\sqrt{T_x}$-limit	
$(\ )_{DD}$	parameter at datum-down position	
$\measuredangle$	action of pilot's power lever	

1. INTRODUCTION

The effectiveness of modern military aircraft with respect to performance capability and maximum usage depends upon the performance reliability of the installed power plant system. This demands not only optimum performance of the propulsion system but also a high degree of repeatability and the implementation of practical methods for monitoring performance based on overall condition of the engine and associated systems. These aspects also influence flight safety and total life cycle cost. Fig. 1 summarises these interconnections.

Performance here refers to the overall characteristics of the installed engine which is a result of actual installed engine characteristics and control system action under given or expected environmental conditions. Prime performance characteristics for a given installation are thrust, fuel consumption, transient behaviour and handling characteristics. Classical steady state performance alone is not sufficient for practical considerations.

Prediction of installed propulsion system performance also plays an important role in pre-flight activities from flight planning up to the engine check immediately before take off. High performance reliability is demanded for V/STOL operations, because small percentages in take-off thrust have a much larger effect on flight performance than for conventional aircraft. These small differences can mean lift-off or not for vertical take-off with maximum payload. Similar requirements apply for manouevering in V/STOL operations and for transition to wing-borne flight.

Fig. 2 shows the main factors determining installed engine performance. For a given set of engine characteristics, i. e. internal condition of the engine, performance is determined by the action of the control system responding to environmental conditions and pilot's commands. There are two loops controlling the engine, the Pilot/Engine Loop and the Control System Loop. Under limiting conditions, e. g. EGT control, the Control

System Loop overrides the Pilot/Engine Loop. This schematic diagramm showing the main sources of influence to installed performance ist the basis for further discussions and will be referred to as the Installed Performance Model. Special attention will be paid to the control system and how it can be used for performance monitoring.

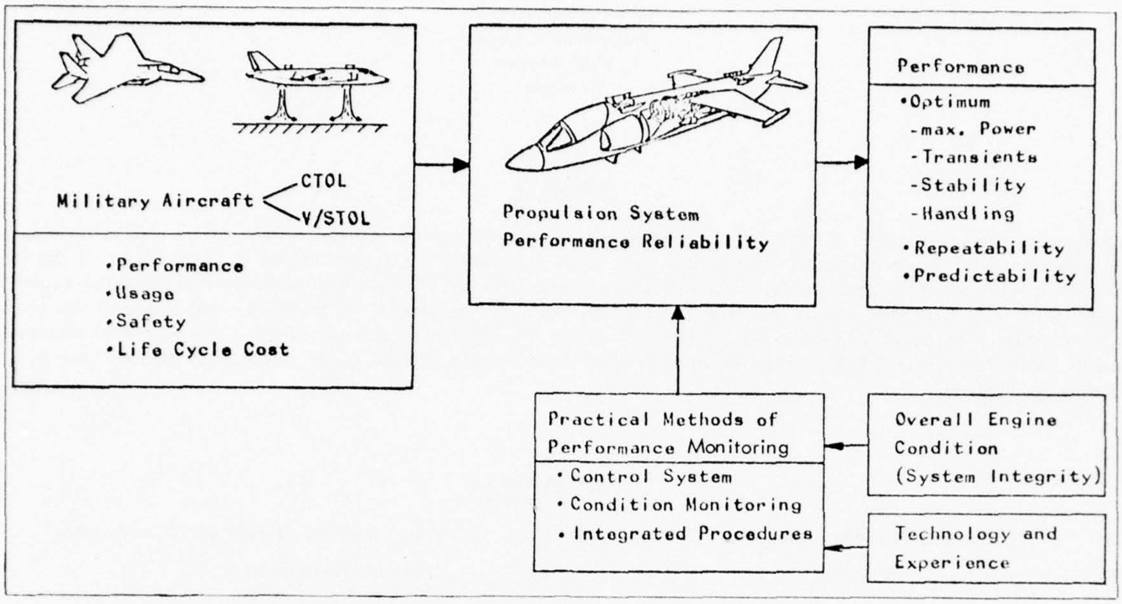


Fig. 1 Performance Reliability

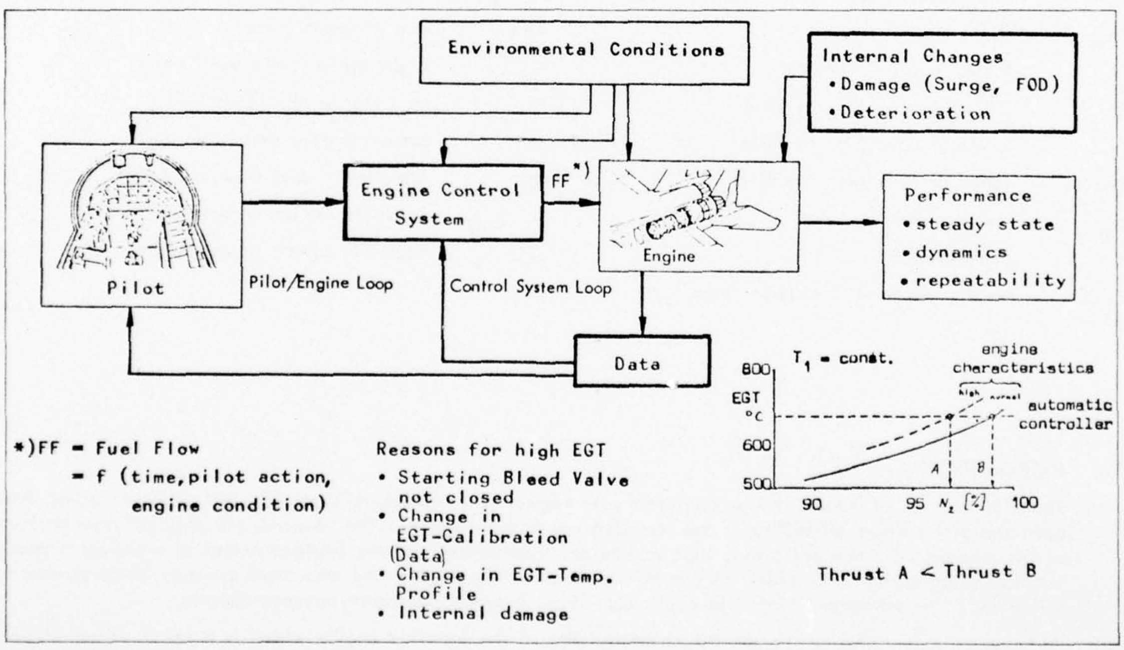
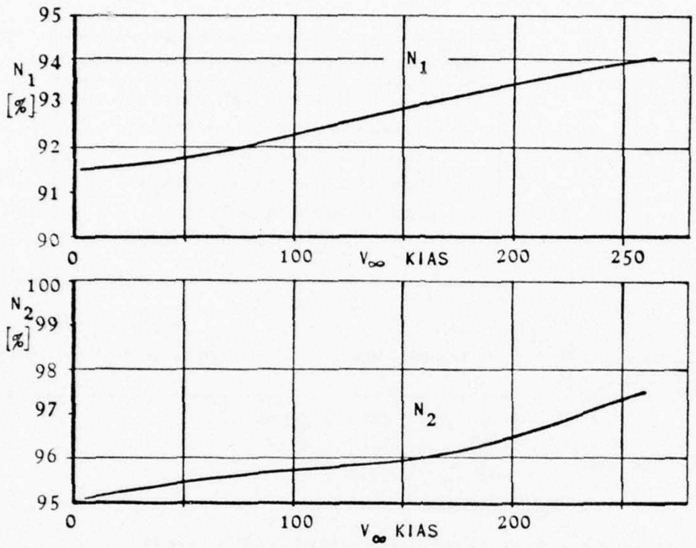


Fig. 2 Factors Determining Installed Engine Performance
(Installed Performance Model)

The characteristics of the engine limiter in the Control System Loop play an important role in the steady state and dynamic working point of the installed engine. Modern military engines may have up to 5 or 6 limiting parameters. Performance characteristics as a result of automatic EGT-limiter action is shown. For a constant EGT-limiter setting changes in the EGT/RPM relationship led to changes in maximum RPM and thus to changes in performance output, i. e. maximum thrust. There are several reasons for changes in the EGT/RPM relationship. Hot running of an engine can be due to failure of a starting bleed valve to close. This can lead to a reduction of take-off thrust on a hot day in the order of 5 % for a non-after-burning military bypass-engine

with bypass-ration 1 : 1. In V/STOL operation using 5 % exess thrust for lift-off this definitely means no lift-off. Maximum performance at other engine limiting parameters and transient behaviour may have different reasons for deviating from the normally expected characteristics but will all lead to changes in the engine running point with a corresponding change in performance. Fig. 3 shows the variation of HP RPM with increasing forward speed of the same engine at $N1/\sqrt{T_x}$-control. Tx is the fan exit total temperature measured with a single probe and experiences greater changes with increasing forward speed that the normal inlet total pressure due to a change in inlet distortion pattern with increasing forward speed. Repeatability of controller characteristics, especially with present day electronic systems, has also to be frequently monitored.



Max. Throttle

N_1 = LP RPM

T_x = Total Temperature at Fan Exit

N_2 = HP RPM

$N_1/\sqrt{T_x}$ = const.

V_∞	N_1	N_2
0	91,5	95,0
200	93,3	96,4
250	93,8	97,3

Fig. 3 Variation of HP RPM with Forward Speed at $N_1/\sqrt{T_x}$ Control

Besides obtaining the control system characteristics integrity of the propulsion system must be checked to be aware of any internal changes due to component change, FOD or excessive deterioration which can cause considerable changes to the basic engine characteristics causing critical variations of installed performance such as

- thrust
- fuel consumption
- handling qualities.

This requires appropriate engine condition monitoring.

Methods for obtaining maximum performance reliability must therefore be built around the following tasks:

- Determine the engine control system characteristics in the installed condition, i. e. in the aircraft with all connections fitted
- Monitor engine condition

Experience gained with current analog-electronic control systems show that once the characteristics have been checked the control system can be used as a tool for quick condition monitoring before flight. The degree and accuracy of condition monitoring actually possible will depered upon the logic built into the control system, the range of flexibility available and implementation. As shown in fig. 4 future military power plant systems will have more variables but will also be equipped with advanced technology control systems.

An efficient methodology for future applications must therefore make the best use of practical experience gained with today's military propulsion systems in order to identify critical areas and take full advantage of future technological capabilities such as digital electronic control systems. This approach is the subject of this paper.

A method will be discussed based on lessons learned in specific practical applications and the possibilities of expected technology in future applications. The basic method was used during flight testing of a V/STOL fighter-type aircraft with a Lift plus Lift/Cruise pupulsion system integrated with the airframe. The aircraft is shown in fig. 5. Attention will be paid to the main engine system which was used for all V/STOL and conventional flight modes. Lift engines were used additionally for V/STOL modes. The engine control system was of the hybrid type.

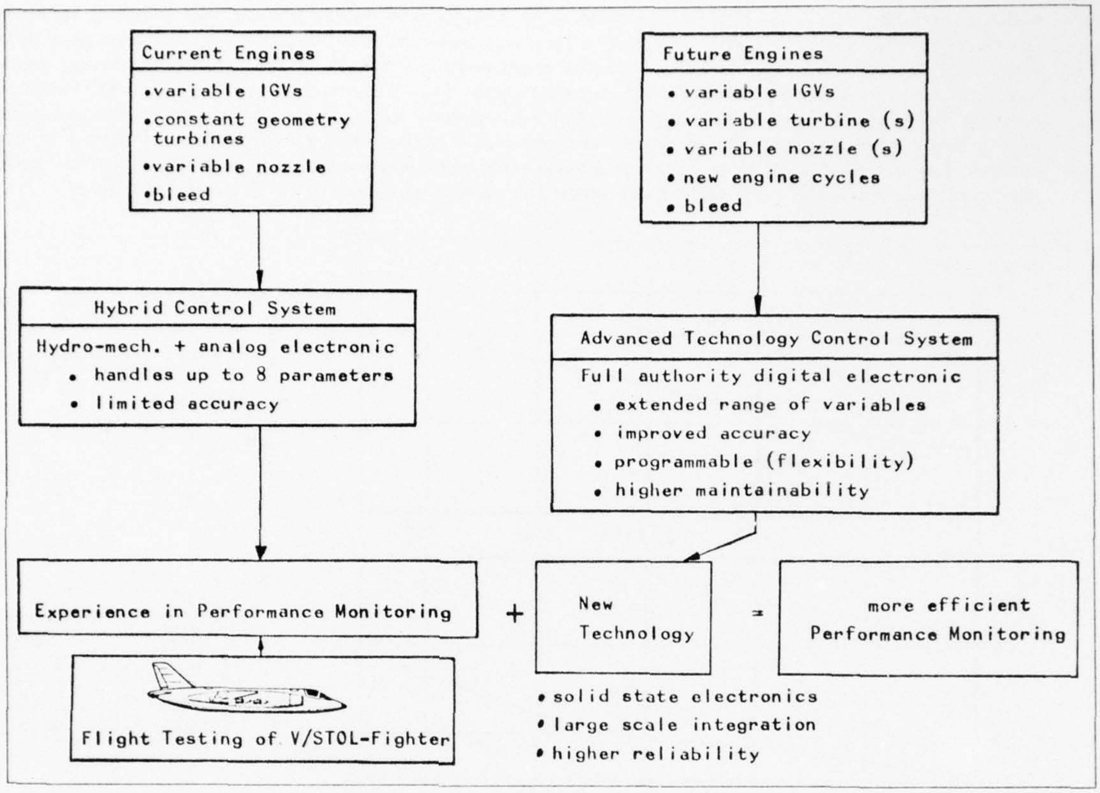


Fig. 4 Performance Monitoring and Technology Standards

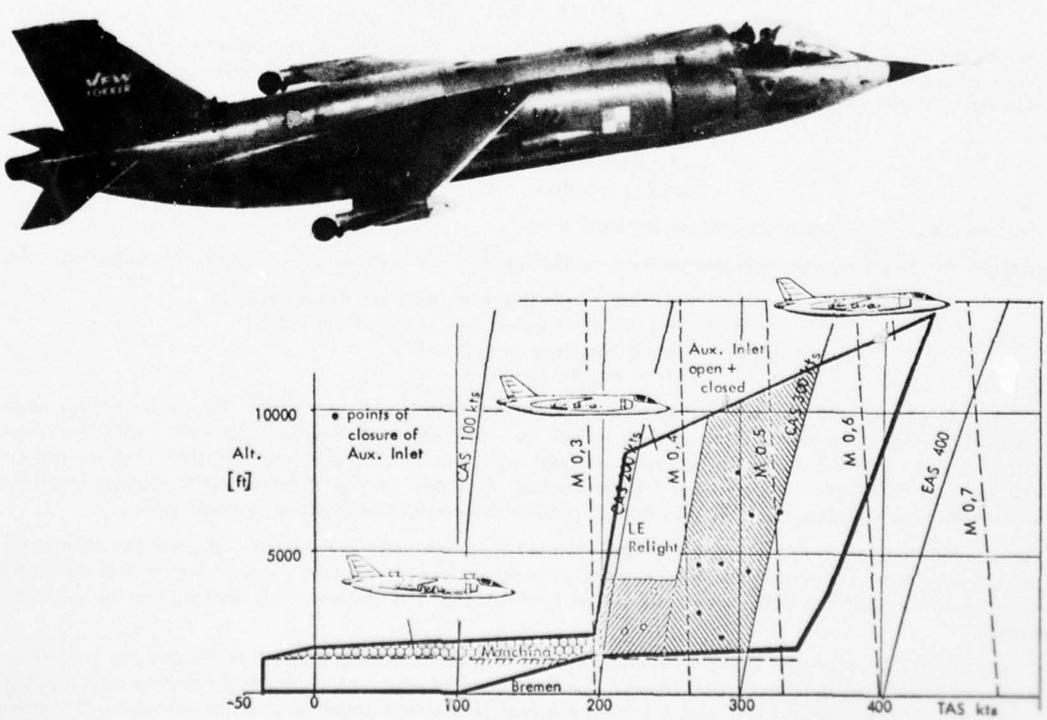


Fig. 5 V/STOL Strike and Reconnaissance Aircraft VAK 191 B

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Experience gained here shows that technology advances correctly applied can significantly improve the efficiency of performance monitoring and thus enhance the reliability of future propulsion systems. This is of special interest to future V/STOL aircraft, which will have very high reliability standards. An outline will be given of future possibilities based on control systems expected with the next generation of military engines. These control systems will be digital electronic making extensive use of solid-state electronics and large-scale integration (LSI). Component testing is already being done with very promising results.

For most propulsion engineers the word "electronics" is still a "taboo". This is mainly due to the "bad" reputation of electronics in the first attempts towards automatic condition monitoring and electrical engine control. The standard of electronics used at that time was not only found to be costly but also less reliable than the system they were supposed to be monitoring. Modern electronics based on solid-state technology and LSI offer fully new dimensions of performance capability, reliability and weight, cost and size savings. Basic technology is proven and will be making further advances, e. g. very large scale integration, to further reduce the number of parts and thereby enhance reliability. The latest pocket calculator models are the most common indicators of the rapid progress. This standard of electronics technology must not be confused with the standard used which led to the "bad" reputation of electronics in engines. Propulsion engineers are getting a new tool in their hands to improve the integration of engine and airframe. It is up to them to make the best use of it.

2. PRACTICAL EXPERIENCE

2.1 Description of Aircraft and Propulsion System

The aircraft is a V/STOL strike and reconnaissance aircraft with a Lift plus Lift/Cruise propulsion system comprising a main engine at the centre of the fuselage and two lift engines, one in front and the other behind the main engine as shown in fig. 6.

The lift engines are ultra-light weight turbojets installed at an inclination of 12.5° to the vertical and have exit doors which can be used as thrust deflection. They are shut down in conventional flight.

The main engine is a twin-spool turbofan with a pair of nozzles each for bypass and hot-nozzle flow. Both nozzle pairs are actuated together over a single air-motor using high-pressure bleed air. Nozzles can be swivelled from 5° to 100° from the horizontal for thrust vectoring. 100° nozzle angle brings a reverse thrust component. This engine thus produces thrust for VTOL and conventional flight. With all three engines running at take-off power the thrust split between main engine and lift engines ist about 1 : 1. Thrust moments of all nozzles about the aircraft C. G. are balanced.

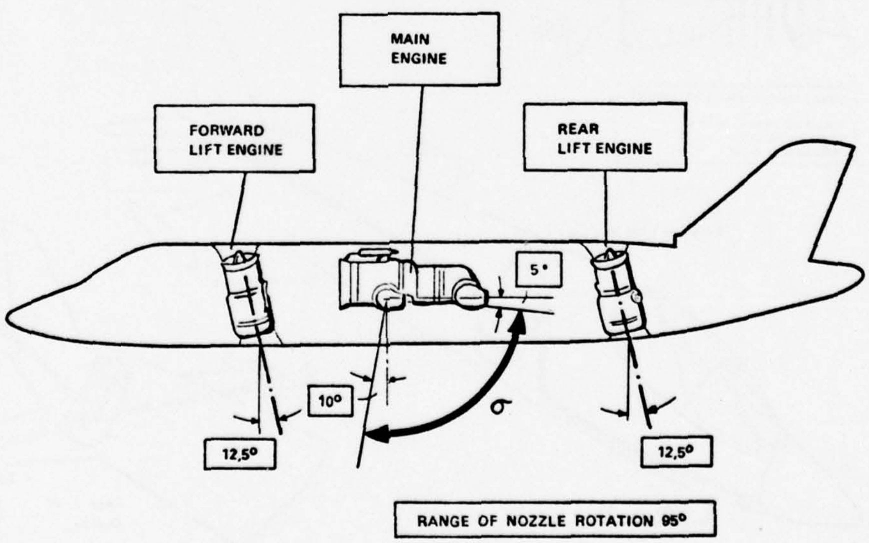


Fig. 6 Propulsion System

For attitude stabilisation and manoeuvering in jetborne flight all engines a capable of producing high amounts of compressor exit bleed. There are no interconnections between the individual bleed systems of each engine. Both lift engines supply bleed to all three axes. In cases of higher control moment demands the main engine bleed system in activited. The main engine supplies all three axes with separate piping and control nozzles. The lift engines produce a continuous bleed of 8 % and in the rear installation intermittent bleed up to 13 %. The main engine can also deliver intermittent high pressure bleed up to 16 % of the gas generator mass flow. The bleed systems of all engines are shown in figs. 7 and 8.

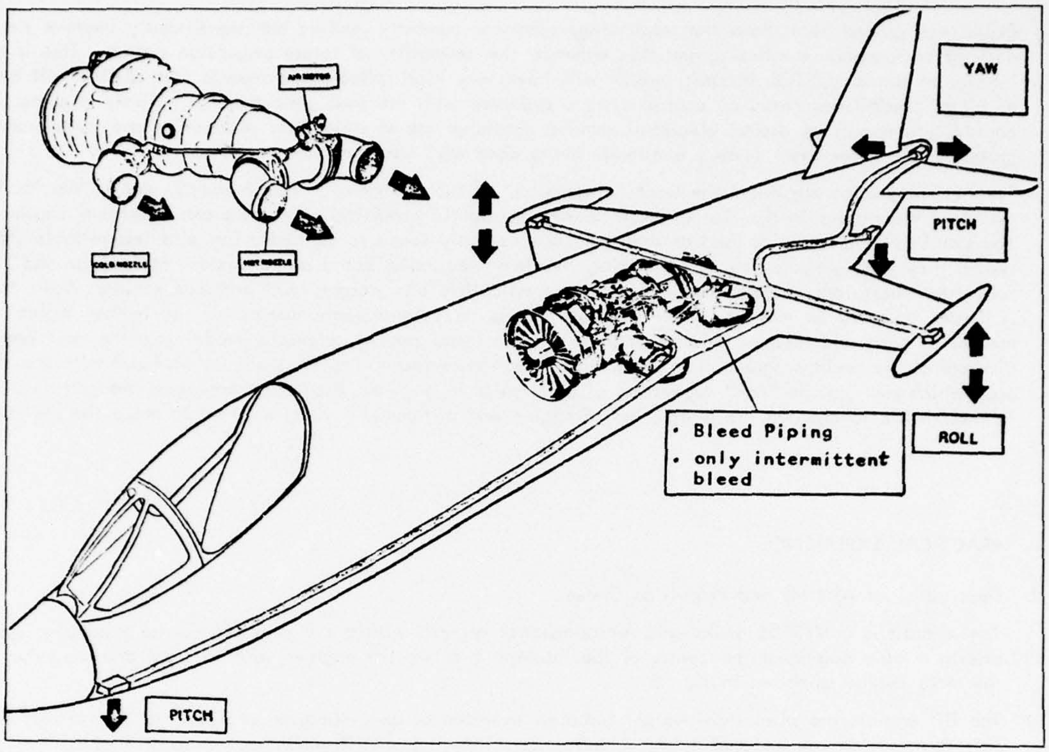


Fig. 7 Main Engine Installation

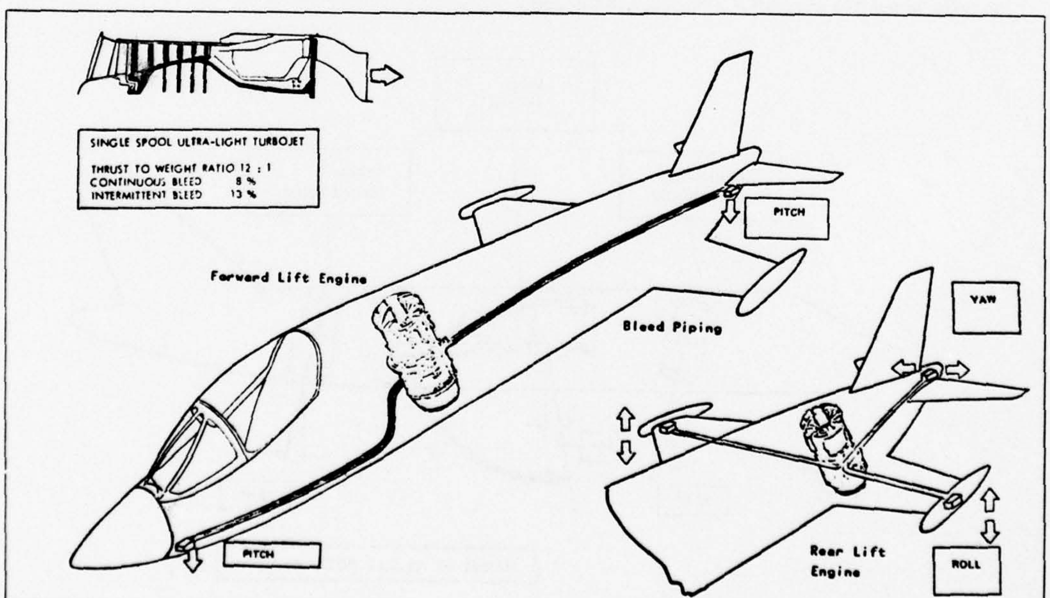


Fig. 8 Installation of Lift Engines

The main engine has a translating cowl intake system which opens an auxiliary side-inlet to improve pressure recovery during V/STOL flight operations. The inlet and outlet doors of the lift engines are closed after lift engine shut down in transition to conventional flight and open again for lift engine light-up to faciliate vertical landing. Inlet and outlet doors are shown in fig. 9.

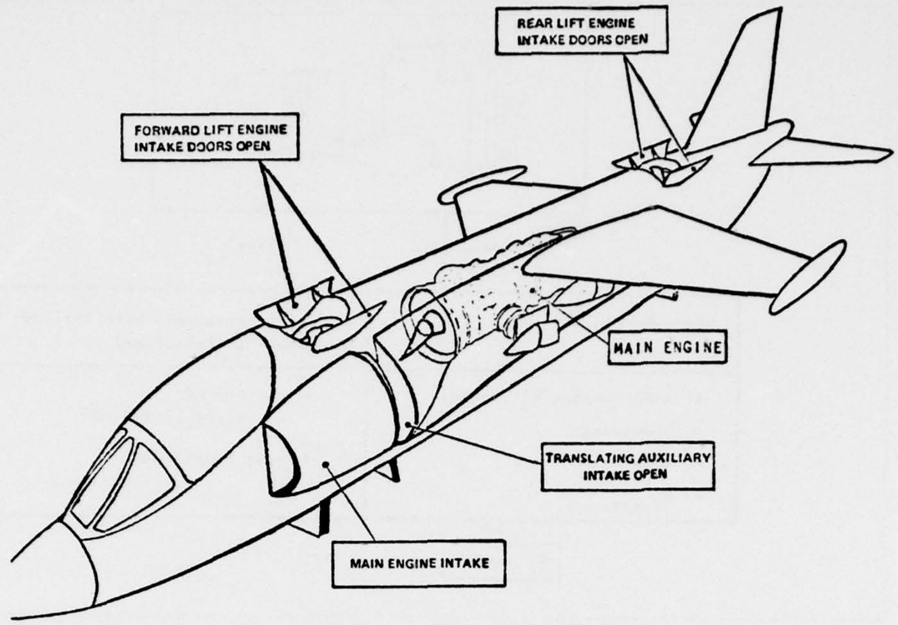


Fig. 9 Intake and Outlets of Propulsion System

Design of the aircraft was for the following main mission:

- o vertical lift off
- o low level flight at high subsonic speeds ($M_\infty > 0.9$)
- o vertical landing

Extensive flight testing has been conducted under both jetborne and aerodynamic fligh conditions. This included lift engine shut down and relight at forward speeds up to 250 kts.

Starting of lift engines on the ground is with bleedair from the main engine. Inflight relighting is possible both by windmilling or with main engine bleed assisstance.

2.2 Main Engine Control System

The main engine has a hybrid control system consisting of a hydro-mechanical unit for isochronous control of HP-RPM N_2 and transient laws and an analog electronic unit for supervisory control of the following limiting parameters:

- o Lp speed N_1
- o corrected LP speed $N_1/\sqrt{T_x}$
- o EGT for V/STOL and conventional modes.

The analog electronic unit also called the "Amplifier" has the provision of special datum-down functions for the above parameters and can be activated from the cockpit. Figs. 10 and 11 show the schematic design of the total control system and the main influence parameters.

The pilots power lever commands N_2. The minimum stop corresponds to Idle speed and the maximum stop the maximum permissible speed N_2 max.. The amplifier always comes into action when any one of the limits is exceeded. Unter these conditions the amplifier has higher authority than the hydro mechanical system and automatically trims down fuel flow to keep the engine at the limit concerned even though the pilot is demanding a higher RPM. Typical action of the amplifier at two limits and a datum-down condition is shown in fig. 12. During flight testing there was the possibility of oberving the output current of the amplifier. This was proportional to the fuel being trimmed off. Action of the amplifier was very rapid.

The amplifier itself is a small metal box mounted on the front engine casing. Presetting of the amplifier can be done in the hangar within less than half a days work including removal and reinstallation.

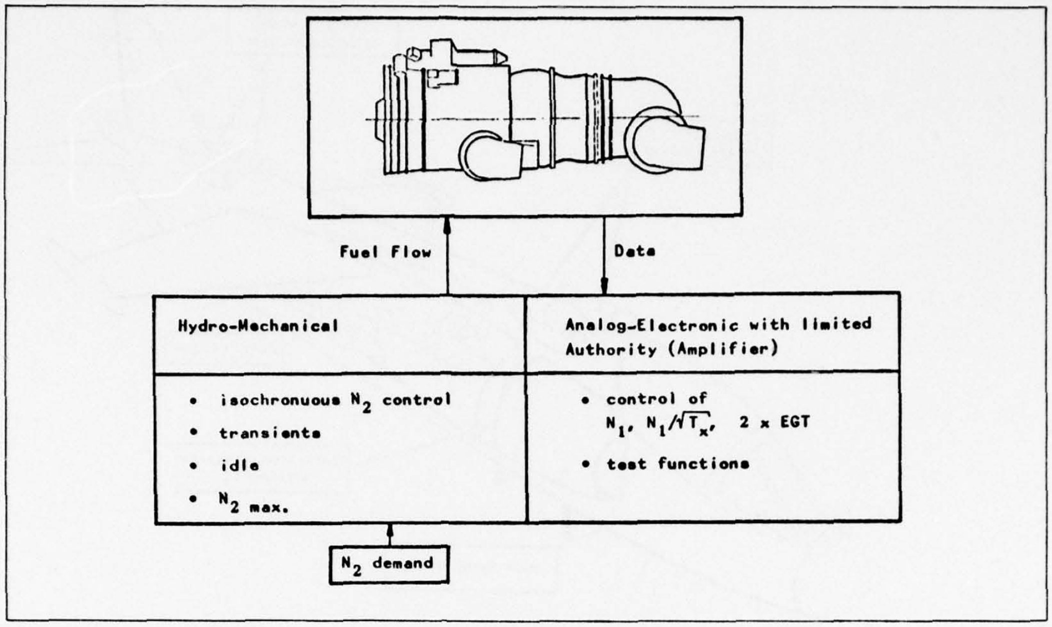


Fig. 10 Main Engine Hybrid Control System

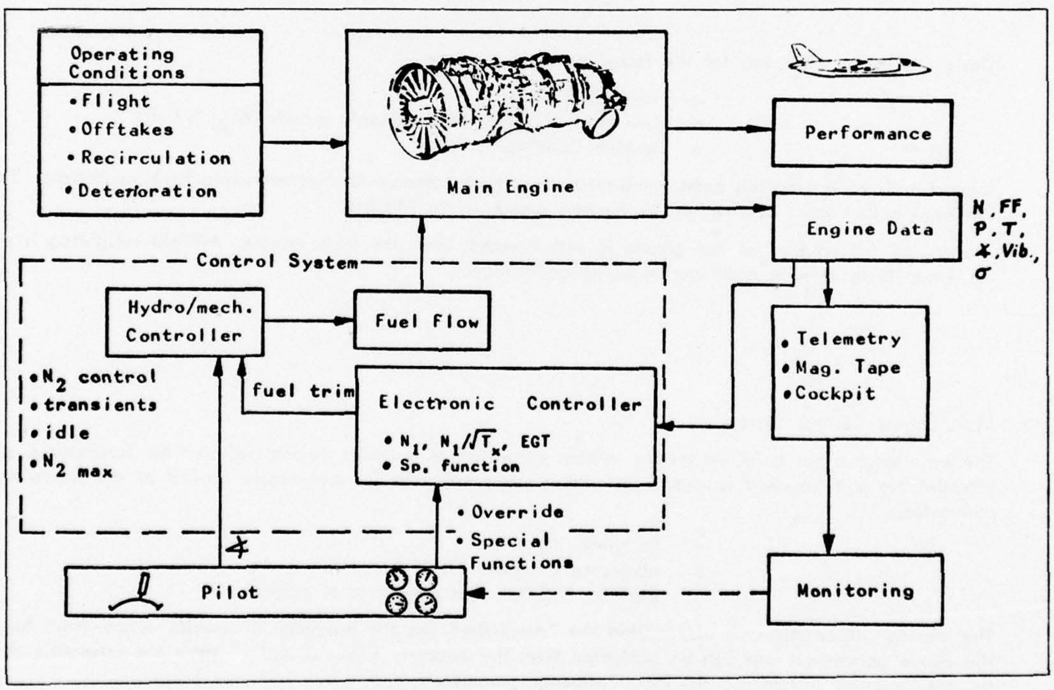


Fig. 11 Schematic Layout aund Influence Parameters

For engine control in the cockpit the pilot has the following instruments:

- o Inlet temperature
- o LP speed N_1
- o HP Speed N_2
- o EGT
- o Fuel Flow
- o Nozzle angle
- o Inlet position (closed/open)
- o Operational mode (V/STOL or Conventional)
  This is defined by the position of the control bleed valve, open or closed

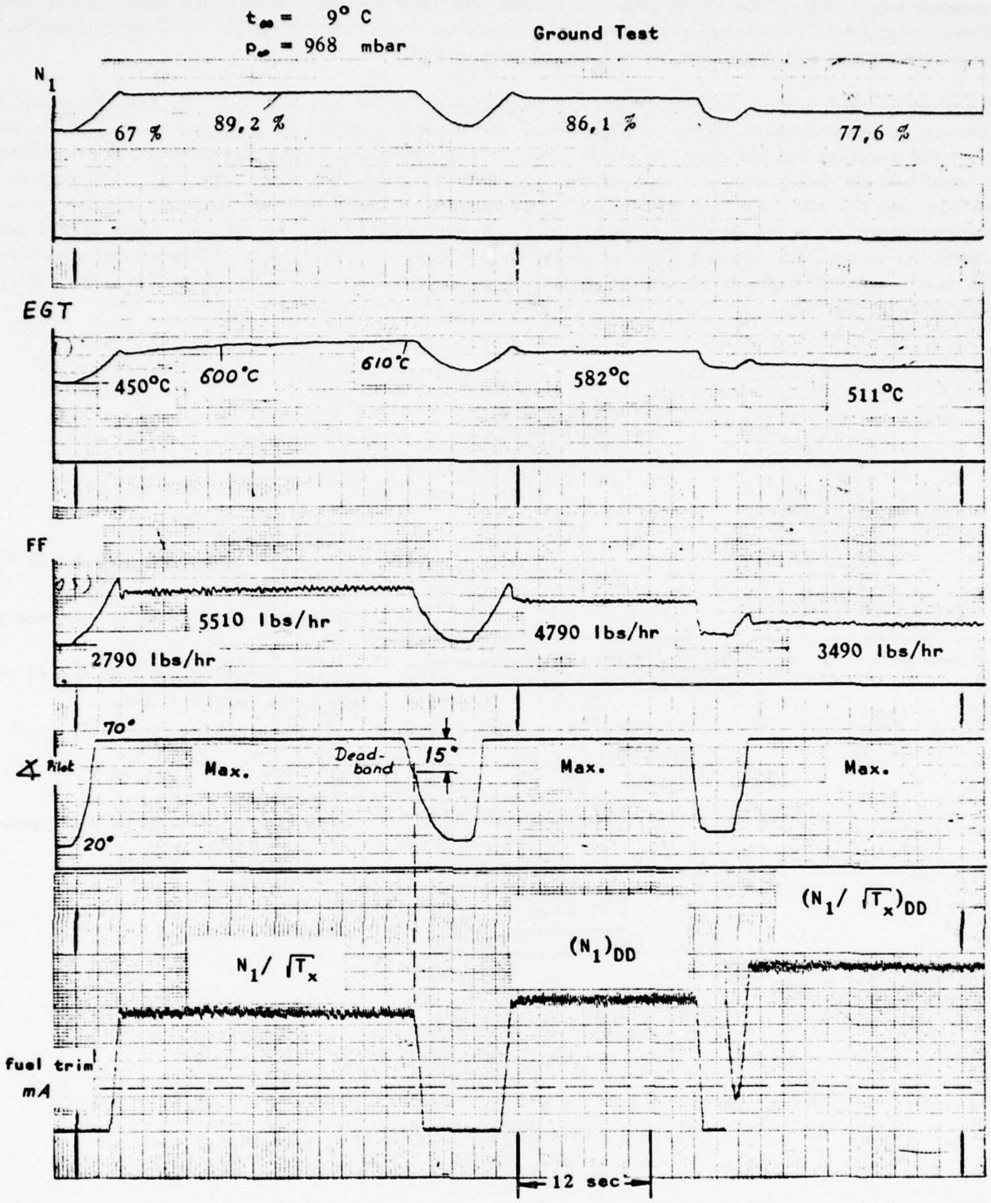


Fig. 12 Engine/Amplifier Operating Characteristics

2.3 Performance Monitoring

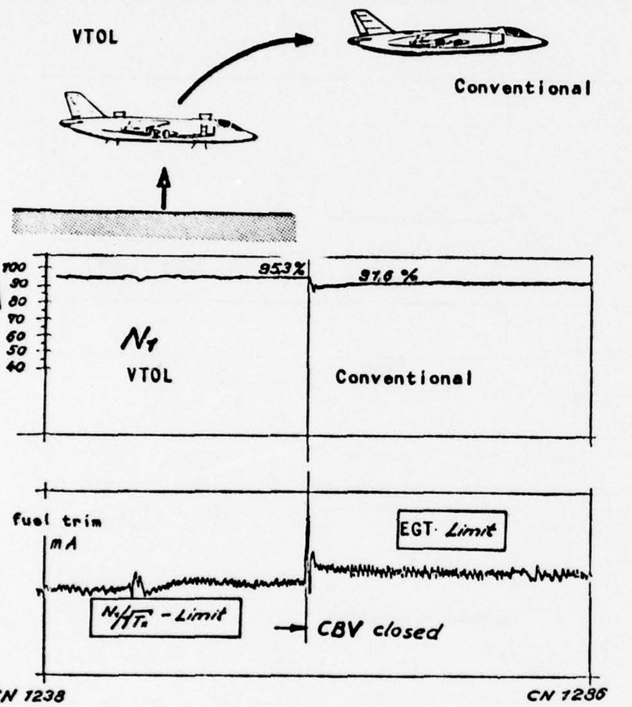
Optimum flight testing demands maximum availability of the aircraft for flight with a minimum of limitations. A further requirement for this aircraft demanded a high standard of reliability for performance predictions, especially with respect to maximum take-off thrust. This directly addressed the interaction of control system and engine characteristics as already shown in the Installed Performance Model. Thus an efficient method of performance and condition monitoring had to be adopted which give quick results and could be applied for decision making during preflight checks and in flight itself. This required best usage of data and technological aids available.

A further objective was to keep pilot workload as low as possible, especially during jetborne operations.

The method employed for the main engine was integrated into the functional check of the analog electronic control system. This will be the basis of further discussions in this paper.

Performance monitoring of the lift engine is a special topic and will not be discussed here. It must however be mentioned that there is a similarity between light-up characteristics during relight in flight and afterburner light-up an conventional prepulsion systems for military aircraft.

The method adopted was an integration of main engineering support activities into the control system check which was an important part of the preflight check out procedure of the main engine system. The procedure consisted of checking the individual amplifier limits in the datum down condition by a series of accelerations from stabilised 85 % N_2 and final acceleration to obtain an indication of the days limit. The purpose was to make sure that the individual electrical circuits corresponding to each limit was working in order to ensure automatic protection of the engine. Although only one limit would apply for take off power it was necessary to check the other limits because flight conditions and automatic control mode change during transition from VTOL to conventional flight can bring the engine up to at last three of the four amplifier limits. This is demonstrated in fig. 13.



VTOL mode

No recirculation, engine is $N_1/\sqrt{T_x}$ or N_1 limited :

Typical condition at maximum power

N_1 = 95,3 %

Recirculation or high bleed off take, engine is EGT limited

Conventional mode

engine is EGT limited, due to lower EGT limit

Typical condition at max. power

N_1 = 91,6 %

Fig. 13 Control Limit Change in Flight

The method used was based on the following facts and is schematically discribed in fig. 14.

o the amplifier was automatically holding a systematic set of stabilised conditions which could build the basic of data for quick and also long term condition monitoring. There is no increase in pilot workload because this is part of a preflight procedure

o once reliability has been established of the relationship between normal and datum down conditions of the limits engine cycling can be minimised by only performing the datum-down checks, or at least the checks only for the limits expected and extrapolating the results to predict maximum performance at normal limiting conditions. This procedure described in fig. 15 saves engine cyclic life, precheck time and fuel.

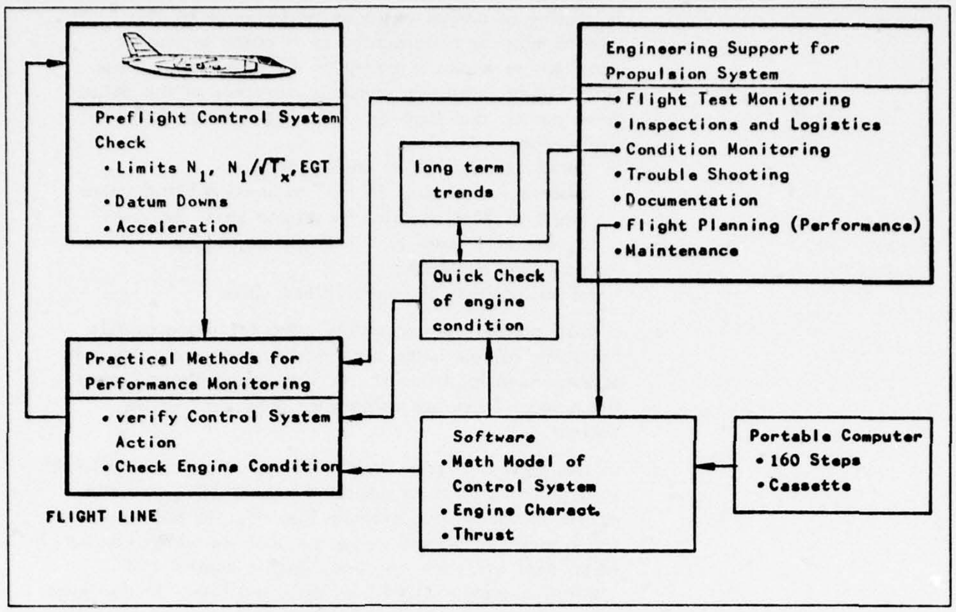
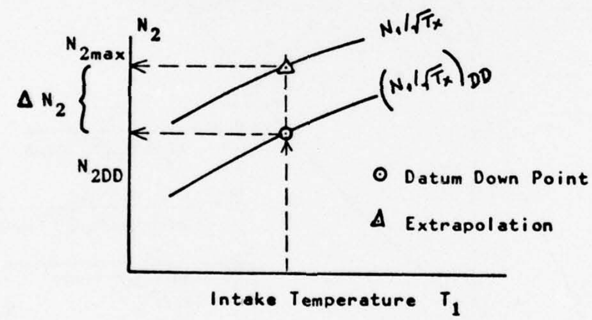


Fig. 14 Development of Methology for Performance Monitoring



ΔN_2 depends upon reliability of acceleration fuel schedule and integrity of the engine and control system

Thrust at N_{2max} depends upon integrity of the engine (is engine o.k.?)

Quick Check of Integrity can be done by

- N_1, N_2, EGT and FF check at idle and datum down points
- observing the following parameters
  a) max. EGT at start
  b) closing of start bleed valve
  c) position of control bleed valve

Fig. 15 Conditions for Extrapolation of Datum Down Data to Predict Max. Performance

Predicting of normal maximum performance by this method must be accompanied by a check of engine condition to ensure integrity of the propulsion system. Data for the condition check is available at the datum down points and from the following characteristics of the engine

- note maximum EGT at engine start
- observe sudden drop in EGT as starting bleed valve closes at RPM peculiar for engine type. An open bleed valve leads to a hot running engine
- note idle conditions
- check position of control bleed valve

o a small portable programmable computer was available for quick calculations. Some modern commercial aircraft already have facilities of this sort called Thrust Rating Computers. These are doing very good work in the cockpit.

o in this particular case there were test engineers available in a ground station to support the pilot. This is useful at the initiation of a program like this. In this way it was possible to manually test out routives which can be taken over by future advanced engine control and monitoring systems as will be discussed later. In the same sense use was also made of digital data generated on-line in the ground station.

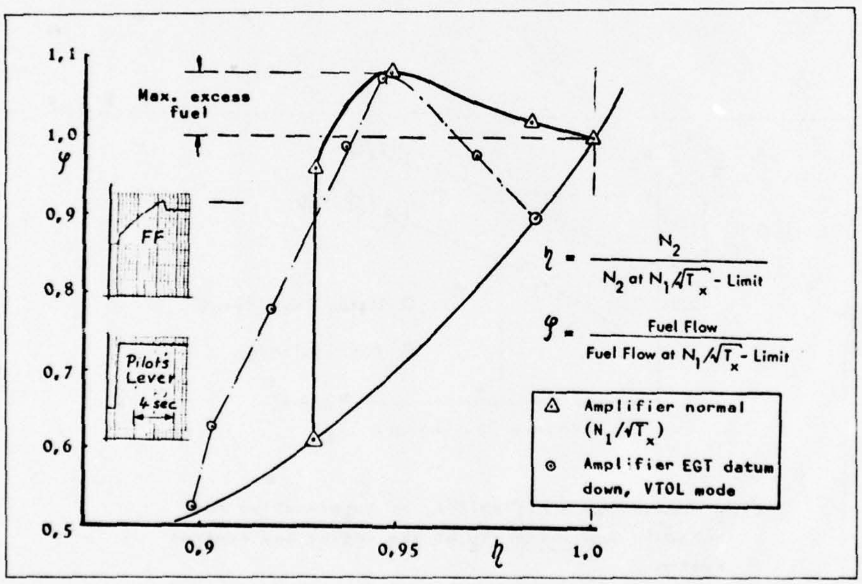


Fig. 16 Analysis of Fuel Flow VS. RPM for Transient Performance

The two main functions of the hydromechanical control system influencing performance are:

o isochronuous control law

N_2 = f (throttle angle)

This is the prime control law of the main engine. Monitoring is quite simple as long as the throttle angle is measured as in the case of the main engine and the lift engines of this experimental V/STOL fighter type aircraft.

- transient characteristics

  Fuel flow = f (N_2, other cycle parameters)

  Good surge free accelerations depend upon optimum excess fuel flow characteristics. Accurate monitoring of these laws is not possible during flight-line operations. For quick check purposes it is only possible to identify an excess fuel flow over the expected value for the finally stabilised conditions. Typical acceleration schedules are shown in fig. 16 for temperature and $N_1/\sqrt{T_x}$ control. Thorough understanding of these fuel laws are necessary for efficient performance monitoring and trouble schooting.

  Current experience with engine installations shows that accuracy of fuel flowmeters is not reliable enough for accurate monitoring and fault diagnosis. Improvements are highly desirable.

  One of the main other "parameters" influencing the excess fuel flow for accelerations is the HP compressor exit pressure P_3.

The engine parameters used for checking out the amplifier and simultaneously doing a quick maximum power and condition check are

$$N_1,\ N_2,\ \text{FF and EGT.}$$

Thes four parameters are also displayed in the cockpit. N_1 and EGT are direct control parameters. The limiting values can easely be determind. In the case of EGT there are two cases defining maximum performance:

1) EGT in VTOL-mode
2) EGT in conventional-mode which is lower than the above.

$N_1/\sqrt{T_x}$ is a further control parameter limiting max. RPM on cold days. This has no direct read out for checking limiter action. Due to its aerothermodynamic connection to $N_1/\sqrt{T_1}$ the N_1 corresponding to this limit is a function of T_1. Additionally all 4 limiting parameters have Datum-Down functions which is simply a reduction of the limiting values by a definite amount. The Datum-Down functions were not used as ratings in flight. They were originally planned for amplifier check out purposes only.

The "pilots" main engine parameter for the check-out procedure was the HP speed N_2. Additional information provided were the two EGT limits. Fig. 17 showes a typical amplifier performance chart. For both the pilot and the flight support crew this type of graphical representation of amplifier and engine performance at defined ratings was very useful for checking repeatability of engine performance and amplifier action. The same characteristics were also programmed into the portable computer.

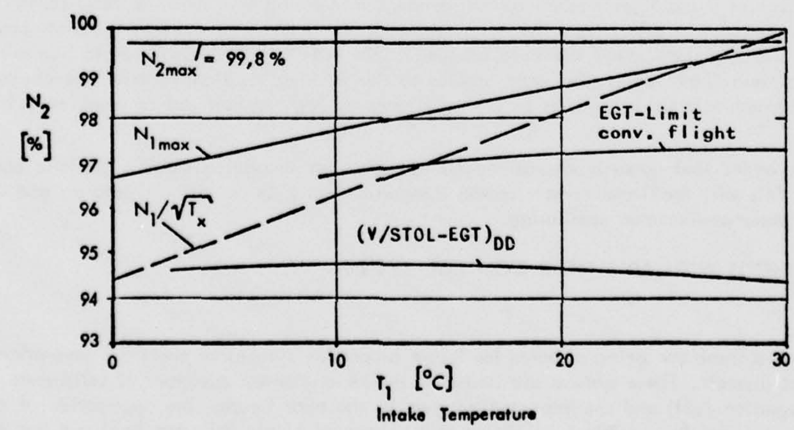


Fig. 17 Typical Amplifier Performance Chart for Installed Engine

The portable computer a Compucorp 326 Scientest with a Compucorp 392 Cassette unit had the following features:

o	weight	7 lbs Compucorp 326
		5 lbs Compucorp 392
o	dimensions	both 14 x 23 x 5 cm
o	battery power	7 V 1,3 Amp.

- o Programmable up to 160 steps per program
- o multiple programs can be stored and used sequentially by addition of a cassette reader.
- o simple programming

The following programs were standard usage

- o characteristics of installed engine N_1, N_2, FF, EGT, P_3
- o amplifier characteristics
- o thrust
- o take-off performance

Running time of the above programs was in the order of a few minutes.

During flight testing on-line digital data was available in the ground station in addition to cockpit information from the pilots instruments. The digital data was used for detailed post-flight analysis and trouble shooting. Post flight condition monitoring included compressor exit pressures and temperatures and limit exceedance of other engine cycle and subsystem parameters. The main points for the post-flight trend analysis were taken from the preflight engine check with the control system. This was a source of systematic data under definied conditions of amplifier control at the individuai limiting parameters. The amplifier was very efficient in keeping at least one parameter constant for obtaining comparable stabilised conditions. The systematic sets of data were entered into the amplifier performance chart for updating and comparison with past results. Consistency of the data points indicated correct behaviour of the control system and unchanged reaction of the engine to limiting parameters. This also meant consistency of installed engine performance.

The individual steps of the method used is discribed in fig. 18. This procedure was part of the preflight check list. In the briefing before the flight the pilot was given the corresponding N_2 values based on a quick run of the portable computer for the days expected values of temperature. The same program generated N_1, EGT and fuel flow predictions for monitoring in the ground station. This standard test procedure or parts of it could also be applied after flight to check the integrity of the propulsion system in the case of suspected deficiencies in flight.

This method proved to be very useful in actual practice. The main advantages were flexibility of operation and the clear separation of engine and control-system characteristics. Engine characteristics were based on the condition check and the control-system on limiting parameters and transients. This also enhanced quick and reliable trouble shooting. Except for accurate handling of the engine to stabilise the initial conditions of 85 % N_2 the rest was an automatic procedure which demanded operation of switches and max. power accels at the correct instances.

Accuracy and location of engine sensors play a dominant role in performance monitoring. As already mentioned improved accuracy of fuel flowmeters is highly desirable. Pressure transducers were checked after each flight for zero-shift. This was done by comparing all absolute pressures with ambient pressure before engine start. Differences to ambient pressure were taken as zero-shift error. In V/STOL aircraft the location of the probe for measuring inlet total temperature must be in a position to give the quickest indication of recirculation which can easily arise during ground runs.

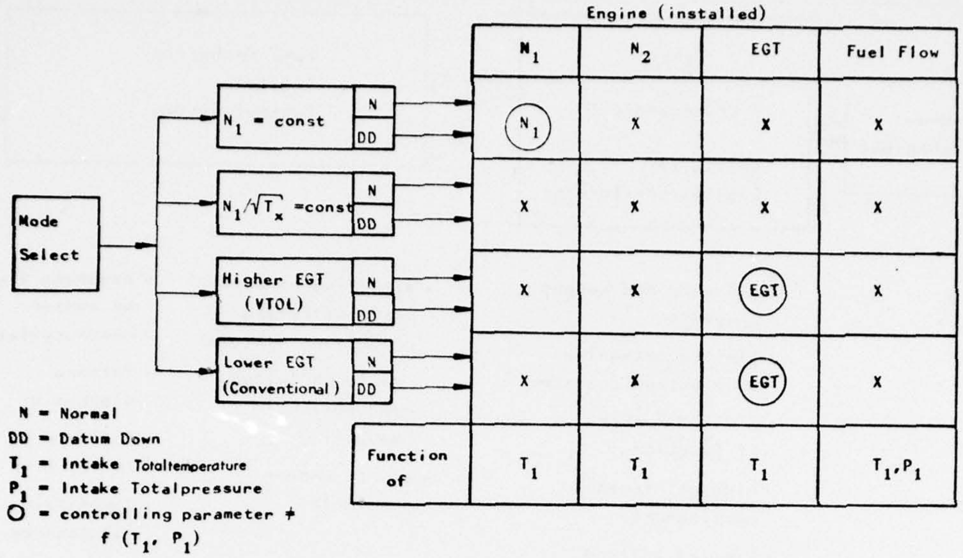
In the development of a practical method for performance monitoring man/machine relationships must be observed. Thus close working with the pilots from the very beginning is necessary to develop practical procedures and make sure that pilots and engineers speak the same language. The man in the cockpit has to live with the propulsion system on "good" and "bad" days. The same applies to the working relationship with the engine manufacturer. All parties concerned must work together as a single team making the best use of experiece, knowledge and technology.

It is also recommended that dynamic engine models for computer simulation contain realistic control system characteristics. This will facilitate correct system integration early in an aircraft program and also enhance procedures for future performance monitoring.

3. PROPOSED METHODS WITH ADVANCED CONTROL SYSTEMS

3.1 Introduction

Advanced control systems are being proposed for future propulsion systems of the next generation of military and commercial aircraft. These systems are based on the revolutionary advances of solid-state electronics and large scale integration (LSI) and the expected progress in the near future. The application of this new technology brings not only drastic reductions in weight and volume of highly efficient hardware but also a reduction in the number of parts through LSI enabling significant improvements in reliability. These facts cannot be overlooked. This standard of electronics must not be confused with past applications which proved less reliable than the system they were designed to monitor. Technology is already available to make the digital control system with full authority a reality. A comparison with current hydro-mechanical and hybrid control systems is shown in fig. 19.



AIRCRAFT 	GROUND STATION engine instruments, digital data, minicomputer
1. ENGINE START - monitor closing of starting bleed valve 2. IDLE - monitor N_1, N_2, EGT and fuel flow	monitor and compute
3. CONTROL SYSTEM AND CONDITION CHECK 3.1. stabilise at 85% N_2 select N_1- datum down slam acceleration to MAX. 3.2. stabilise at 85% N_2 select $N_1/\sqrt{T_x}$ - datum down slam acceleration to MAX. N_2 = 90,5 % 3.3. stabilise at 85% N_2 select EGT- datum down slam acceleration to MAX. N_2 = 92,5 % 3.4. datum downs OFF 3.5. stabilise at 85% N_2 slam acceleration to MAX. N_2 = 97,5 % observe EGT - limit go to IDLE	T_1 / N_2 / EGT predicted 15 / 90,5 / 560 test 16 / 90,2 / 570 predicted 15 / 92,5 / $\boxed{580}$ test 15 / 92 / 583 predicted 15 / 97,5 / 680 test 14 / 97,6 / 690
4. BLEED CHECK 5. LIFT ENGINES START	o check amplifier limits o check condition
Tolerances: ΔN_1 = 1 %, ΔN_2 = 0,5 % Δ EGT = 10° C	o compare with predicted max. conditions o data for trending and trouble shooting

Fig. 18 Engine Preflight Checkout with Integrated Performance Monitoring Procedures

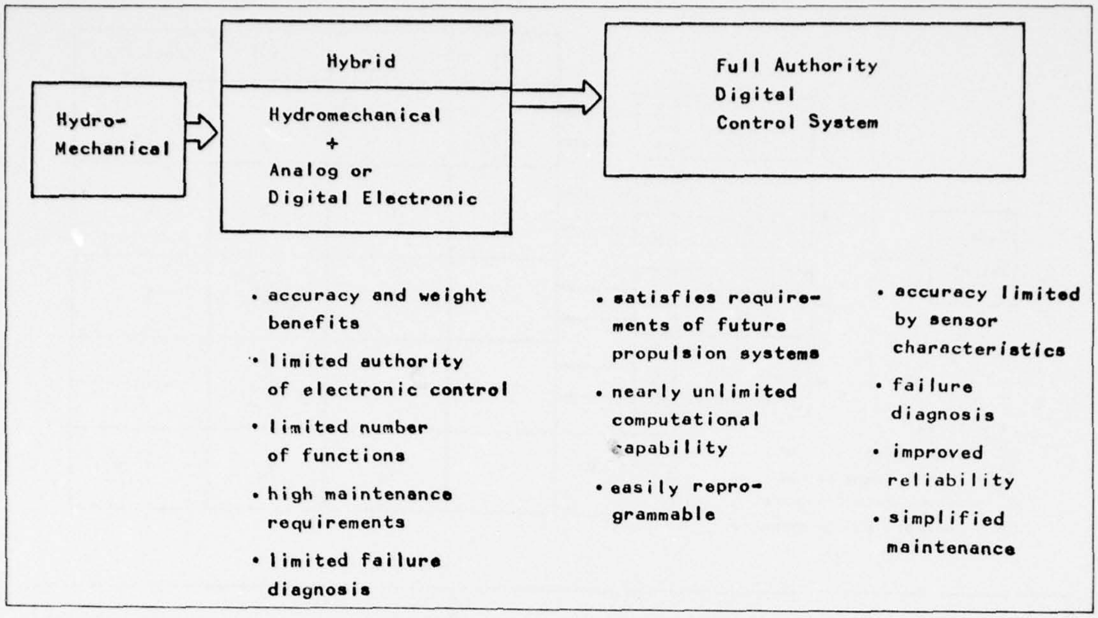


Fig. 19 Comparison of Control Systems

Thus future propulsion systems with this standard of advanced control systems will be offering a new range system flexibility and functional capabilities to improve the integration of engine and airframe and expand the operational possibilities of future aircraft. Substantial improvements in life cycle cost and performance reliability will result.

The key to reliability is to reduce the number of parts. This is enabled by solid-state electronics incorporating more and more circuits and logic design per chip through large scale integration. Very large scale integration with low power requirements are expected as further developments.

This technology is already being exploited for the development of new digital control concepts for future military engines, especially those of the variable cycle type, which will be more complex and will be imposing additional functional requirements to the control systems. A summary of the main efforts in fig. 20 shows that progress is being made and future engines will be equipped with these control systems. The advantage of this technology has already been recognised by the manufacturers of commercial engines, who together with specific airframe manufactures already have their development programs underway.

PROGRAM	INVESTIGATORS	STATUS
FADEC = Full Authority Digital Electronic Control	US Navy	1) Demonstrator tests on PW TF 30-P-412 as slave engine 2) Application studies for future VCE
IPCS = Integrated Propulsion Control System	Boeing/Pratt & Whitney/ Honeywell	1) Build and ground test a digital control system 2) Flight testing
Research & Development on digital engine control	England Germany USA Soviet Union x)	
microprocessor and associated accessories	electronics industry automobile industry	1) Very large scale integration, less parts 2) Higher reliability, lower costs 3) More logic per chip 4) Lower power requirements 5) Drastic size reductions (factor of 10 every 7 years)

x) Lotarev D36 turbofan in YAK 42 with digital control system

Fig. 20 Current Efforts on Engine Digital Control

The possibilities offered here offer more efficient integration of control system characteristics and engine condition monitoring operations to improve the performance reliability of future power plant systems. To demonstrate practical possibilities a method of performance monitoring during engine pre-flight check will be developed based on experience gained with a similar procedure for the main engine system of a V/STOL fighter type aircraft with a hybrid control system as already described. The same philosophy can be applied to applications with more complex power plants. The basis here will be an appropriate Installed Performance Model. For time and engine cyclic life savings maximum use must be made of built in test logic for self monitoring of control system characteristics and fault accomodation.

3.2 Methodology

The advanced control system will be able to perform the functions of both the hydromechanical and the analog electronic control system . Flexibility of the system will also make it capable of taking over additional functions. This new situation in the control area is displayed in fig. 21.

The advanced control system will have a substantival memory and modern logic elements for quick computation. Thus it will be capable of performing the following tasks, which were not possible with the hybrid system:

- o self check of control system compare limiting engine parameter N_1, $N_1/\sqrt{T_x}$ or EGT with the commanded limit
- o self check of $N_{2\,max.}$ at the above limits
- o introduction of N_2 datum down at 85 % for automatic setting of initial conditions for control system and engine check out.
- o compute acceleration time and check fuel schedules

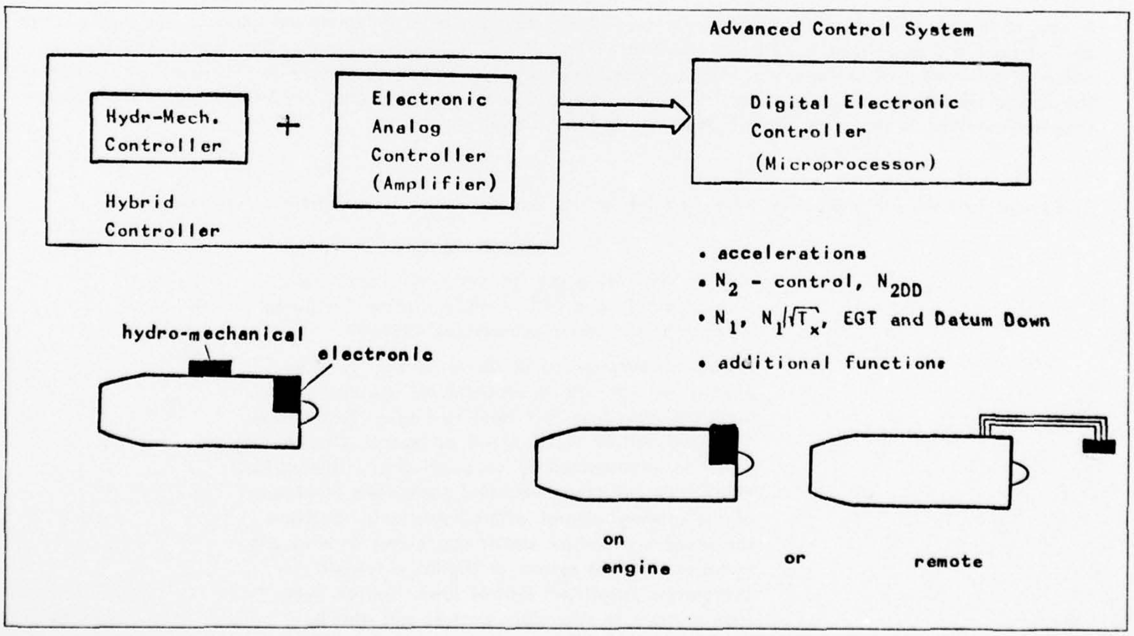


Fig. 21 Situation with advanced Control System

- o perform engine condition check
  Engine characteristics for N_1, N_2, $N_1/\sqrt{T_x}$, EGT and FF can be stored in controller memory and computed for limiting conditions. Thus the advanced controller will be able to execute the calculations and comparison done in the ground station with the aid of a portable minicomputer. If the engine preflight check does not include the acceleration to maximum RPM to see if the engine was going to meet expected maximum performance for that day, then by analysis of the transient characteristics of the datum down data and a condition check the ability of the engine to meet maximum performance can be automatically investigated by the system and the result displayed to the pilot in the form of
  a) Go / No Go or
  b) instruction to perform the full acceleration in cases of uncertainly .

  Due to the greater accuracy of digital data and the extanded computing facibility the engine condition check can be expanded to include compressor exit pressures. The results these computations can be recorded on the flight data recorder and thereby save postflight computing time on the ground.

- o condition monitoring at stabilised conditions in flight is possible. The selection can be manual or automatic. Real time monitoring in flight has a significant effect on flight safety. There are a number of cases when this facility is useful, e. g. when an unexpected surge occurs during flight. The pilot first looks at his warning panel and then at the engine instruments. Even if all appears to be well additional information from an engine condition check will be very useful in the decision making process for the pilot to either continue his mission or return.

A new check out procedure making optimum use of the capabilities of an advanced control system to monitor control laws and engine characteristics is shown in fig. 22. This is a first step in time saving and more accurate monitoring of performance. The advanced controller has taken over tasks which were not possible with the hybrid system and which required the manual operation of a portable computer with inputs either from instrument readings or real time digital data in a ground station.

Additional features for improving the monitoring of performance and system reliability will be:

- o display of limiting N_2 for the pilot similar to the display of N_1 and EPR in Thrust Rating Computers in some of the latest commercial aircraft
- o failure accommodation as shown in fig. 23. The primary control will incorporate all the desird functions including Self Healing Logic. This allows for signal failure of a control parameter. The missing signal is compensated by calculation of a theoretical value from the other measured parameters. Fall out of the primary control will automatically activate the secondary control which can either be a simple hydro-mechanical system or digital electronic incorporating simplified control laws. Switch over from primary to secondary control can also be facilitated by manual override.

COCKPIT	CONTROL SYSTEM (Microprocessor)
1. PART POWER CHECK	
1.1. Engine throttle to IDLE ∡	
N_2- datum down ON	
Slow acceleration to MAX N_2= 85 % ∡	controller monitors engine condition with N_1, N_2, EGT, FF and P_3.
N_1- datum down SELECT	
N_2- datum down OFF $N_1 = (N_{1\,max})_{DD}$	Transient check
1.2. N_2- datum down ON N_2= 85 %	Diagnostic results on tape
$N_1/\sqrt{T_x}$ - datum down SELECT	
N_2- datum down OFF N_2=predicted	
1.3. N_2- datum down ON N_2= 85 %	
EGT- datum down SELECT	Unsatisfactory condition check leads to REPEAT display in cockpit
N_2- datum down OFF EGT = $(EGT)_{DD}$	
1.4. Engine throttle to IDLE ∡	
All datum downs OFF	
2. FULL POWER CHECK	
N_2- datum down ON	
Slow acceleration to MAX ∡	controller checks condition, transients and maximum performance
N_2- datum down OFF $N_{2\,max}$	
Engine throttle to IDLE ∡	

Fig. 22 Integrated Preflight Check with Advanced Controls

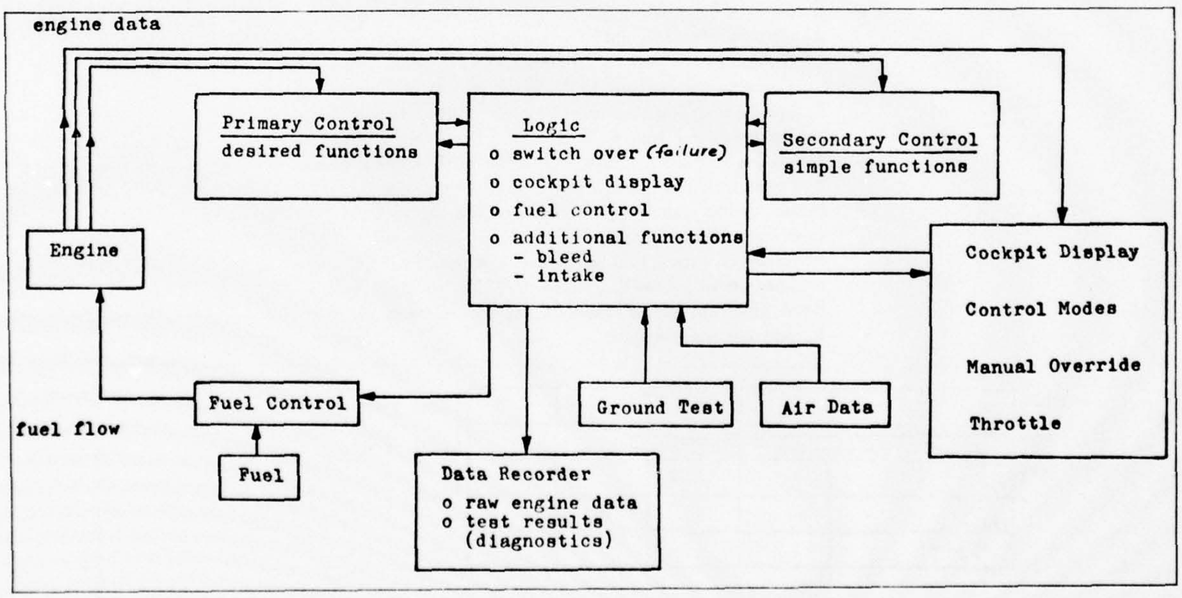
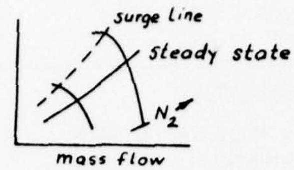


Fig. 23 Schematic Layout of Advanced Control System

Performance reliability can be further improved by integration of the control logic of the following components of the installed propulsion system with the primary digital control system on the basis of a coordinated control concept for monitoring not only the basic engine and its control system but also other subsystems influencing installed engine performance as shown in fig. 24



- Surge margin control
- Coordinated intake control

- Elimination of dead-bands in pilot throttle
- Automatic updating of sensor calibration
- Minimise ground running of new engines by improved engine trimming
- Improved diagnostic capabilities to minimise maintenance and maximise usage of engine and aircraft

Fig. 24 Further Application of advanced Control Systems using Extensive Data Processing

o bleed schedules
a) Compressor bleed-off for surge margin control
b) Compressor bleed for attitude stabilisation and lift engine starting in V/STOL aircraft

Engine bleed degrades performance especially at maximum power and EGT control. Unreliable bleed schedules not only cause excessive performance losses but also stability problems. Opening and closing of bleed valves can be detected by changes in EGT and compressor discharge pressure at stabilised conditions. Fig. 25 shows these changes for lift engine starting bleed. Future V/STOL aircraft will need accurate bleed bookkeeping for reliable performance monitoring.

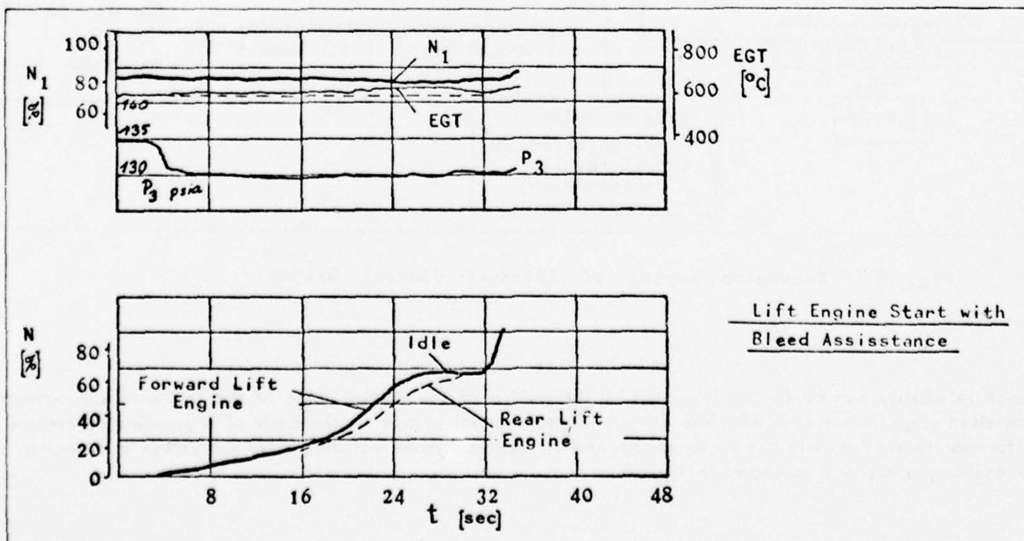


Fig. 25 Influence of Bleed on Engine Characteristics

- o control of variable intake
  The simplest is an intake with a translating cowl type of auxiliary inlet for optimum pressure recovery at take off and law speed flight. Accurate opening and closing of the auxilary inlet can be controlled by integrating the intake logic into the advanced control system. A similar philosophy can also be adapted for supersonic inlets with variable ramps.

- o elimination of dead-bands in the pilots power lever and improve power setting methods
  This is a problem common to mechanical power plant controls when the pilot demands maximum power and the limiter comes in keeping a limiting parameter constant. This leads to a dead-band at the upper end of the pilots lever within which there is no response from the engine to movements of the pilots lever. Dead-bands are highly undesirable from the standpoint of engine handling. The use of electrical controls together with future advanced control systems will facilitate the elimination of dead-bands and improvements to power setting methods. The cockpit workload will also be reduced.

- o minimise maintenance and maximise usage of engine and aircraft by accomplishing the following tasks is real time:
  - calculate turbine creep life usage
  - calculate usage of cyclic life
  - monitor engine running time
  - in additions to condtion monitoring apply diagnostic routines for fault identification and indicate corrective action in the cockpit
  - elimination of ground trim runs after new engine installation

- o optimisation of power management in flight by integration of aircraft and engine controls using criteria based on real time engine performance data.

- o new range of flexibility for the treatment of problems that arise during initial engine and airframe development.

- o control of engine sensors, e. g. automatic compensation of pressure transducer drift.

3.3 Development Requirements

The proposed preflight check-out procedure taking advantage of the abilities of an advanced engine controller on the basis of a digital microprocessor and further examples of extended application show the wide range of possibilities for improving installed performance and overall system reliability of future powerplant systems. The central microprocessor with its high data storage capacity is capable of efficiently performing multiple operations from control-law check-out up to different stages of engine health diagnosis.

Allthough the basic technology is available and has already demonstrated its ability as an engine control tool on the test bed and in at least one case in flight (F 111 E) there is still a wide spectrum of research and development necessary for determining the best methods for applying this new technology to improve the overall performance and system reliability of military power plant systems. The development of methods for practical applications must be based on comprehensive trade studies considering the following aspects:

- o mission
- o vehicle configuration
- o integration with airframe
- o control system reliability
- o procurement cost
- o life cycle cost of propulsion system
- o maintainability
- o dispatchability
- o other parameters affected by day to day practical application.

Propulsion engineers in the engine and airframe industry must acquire the know-how necessary to handle this new technology and solve interface problems. This requires experimental testing of prototype and production standard hardware on the ground and in flight in the form of joint efforts by engine and airframe manufacturers. These efforts must start today. Propulsion engineers must gradually lay off the "tradional" apprehension to electronics. We are getting a new and usefull tool in our hands and must make the best use of it. Avionics engineers have made better progress in this respect.

A further objective is the need for standardisation of components for "propulsion-electronics" and test procedures. This will improve economics and reliability.

One very important interface item between the engine and all systems using engine data for control and monitoring purposes are the engine sensors. Optimum results from the application of methods making best use of system logic and data handling capability for improving performance monitoring will require better instrumentation from the standpoint of accuracy, response time and reliability. Future sensor requirements are listed in fig. 26.

Sensor	Accuracy current (%)	Accuracy required (%)	Response Time (sec)
Temperature	± 0,5	± 0,1	0,1
Pressure	± 0,5	± 0,1	0,02
Fuel Flow	> 1,0	± 0,5	0,05
Speed	± 0,5	± 0,1	0,02
Power Lever Angle	≈ 3	± 1	0,02

Fig. 26 Sensor Requirements

CONCLUSIONS

Installed engine behaviour depends upon engine condition and the characteristics of the engine control system. Thus efficient performance monitoring must consist of a control system and engine condition check.

Current automatic multi-parameter limiters in the form of analog electronic systems with limited authority require defined preflight checkout procedures to confirm limiter action for flight. Datum down functions of the individual limiting signals can be used for this purpose at part and maximum power. This procedure delivers a systematic set of quasi steady state engine data for performance monitoring both in flight line as well as after flight for trend analysis, detailed diagnostics and trouble shooting.

Use of a portable minicomputer in a ground station with communication to the pilot to facilitate real time overall performance monitoring was a technique used to develop an intergrated procedure for obtaining an early indication of engine condition and performance capability. This integrated methodology was applied to the performance monitoring of the propulsion system of a V/STOL fighter demanding high standards of performance reliability for jetborne performance. Practical experience in flight test demonstrated the ability of this approach as a quick-working and flexible tool making efficient use of quasi steady-state engine data generated during preflight checkout for the purpose of quick condition monitoring and to give the pilot an indication of performance capability. Similar integrated methods can also be applied to other complex military power plants. A suitable bord computer can take over the necessary calculations and thus make the whole system integral with the airframe and indepandet of a ground station. Commercial aircraft already use power setting computers.

Advanced digital electronic engine control systems based on microprocessor technology and large scale integration will be able to perform the above procedures more efficiently and also have the capability of taking on additional functions. These functions go from control system check-out up to power management in flight.

Further development and research as shown in fig. 27 is necessary to provide a solid basis for the efficient application of this very promising new technology. The very high standards of reliability required for full authority control systems demand substantial experimental testing on real engines in actual environments of temperature and vibration. This requires coordinated effects from airframe, engine and control system manufacturers.

Optimum use of current systems and know-how	+	new technology • established • ground tested • flight tested • standardised	=	practical methods for improving performance, usage and life cycle cost of future propulsion systems

- Trade studies for application of optimum standard of technology.
- Establish standards for test and application of "Propulsion Electronics". Standardise and define modular components
- Conduct hardware testing on the ground and in flight. Simulate failure modes

Coordinated development efforts supported by
- engine industry
- airframe industry
- systems manufacturers

working together.

Fig. 27 Development Requirements

Development in this expanding sector of "Propulsion Electronics" will benefit from a standardisation of the main components of this new technology for future power plants based on modular concepts. There should not be too many totaly different "black boxes" each with it's own "secret" known only to certain individuals.

It is proposed that a specialist working group within AGARD/PEP pay special attention to the relevant aspects of advanced engine control systems for optimum integration with the airframe to improve performance, reliability and life cycle cost of future military aircraft. The subject of engine condition monitoring must be one of the main topics of this working group.

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PRELIMINARY RESULTS OF USAF EXPERIENCE WITH ENGINE MONITORING AND DIAGNOSTICS

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SUMMARY: The United States Air Force is currently conducting a formal flight test evaluation of an Engine Health Monitoring System (EHMS) on ten of the Air Training Command's T-38 supersonic aircraft. The system consists of engine sensors, an airframe mounted data processing unit (DPU), and a ground based diagnostic display unit (DDU). The sensors continuously monitor some 24 parameters including an EGT, RPM, fuel flow and EPR; however, data is only recorded if the pilot wishes or if a gate in the DPU is triggered.

During the flight evaluation program, an extensive amount of data is being collected on the ten aircraft and 20 instrumental engines, as well as on a control group of an additional 20 engines. As the test progresses, a series of logistic comparisons will be conducted to determine the extent to which an EHMS system can -

- fault isolate to module/LRU level
- reduce spare engine and piece part requirements
- reduce fuel consumption
- improve flight safety
- mechanize data collection
- reduce scheduled engine removals
- trend performance data
  and in general
- increase aircraft availability while reducing logistics support costs

Since this is the first controlled experiment in the USAF to determine the evidence of a potential for Engine Health Monitoring Systems, the results are assumed to be general interest.

The United States Air Force is actively involved in the evaluation and expansion of various engine diagnostic techniques and engine monitoring systems for the purpose of developing an aircraft engine on-condition maintenance program. The F100 (F-15/F-16), F101 (B-1) and TF34 (A-10) engines are being added to the Air Force aircraft engine inventory under this on-condition maintenance (OCM) concept. These engines will be supported by conditional maintenance procedures that do not require mandatory field maintenance involving periodic inspections (P/E) and depot maintenance controlled by maximum operating times (MOT). Application of the conditional maintenance concept varies in specifics as applied to each engine but all engines will require some degree of diagnostic capability to realize the full benefits of OCM. The more well established diagnostic procedures of Spectrometric Oil Analysis Program (SOAP) sampling, Nondestructive Inspection (NDI) and borescope will continue to be used on engines maintained by the OCM concept. However, additional diagnostic capabilities, that can be satisfied by monitoring selected engine parameters in flight are required.

The current maintenance concept for aircraft jet engines is based upon scheduled intervals for field level maintenance actions (inspection, remove, replace, and repair) and depot level maintenance (complete engine overhaul, including modifications). These intervals are expressed in total engine operating hours and are established, and revised, based upon the condition of critical components observed during maintenance. If reasons for unscheduled engine removals or parts consumption factors show significant changes, the field maintenance interval (periodic inspection) is adjusted as is the depot repair interval (maximum operating time).

Most engines are returned for depot repair for reasons other than reaching their maximum operating time. Engines are rejected for in-flight malfunctions (stalls, flameouts, surges, vibration, etc) and component failures of rotating parts (turbine/compressor blades and disks) and critical static parts (combustion chambers, turbine nozzles). These type of malfunctions cause engine removal and subsequent field testing. Frequent engine removals require more spare engines and field level repair consumes maintenance manhours and spare parts. Ground operational checks of malfunctioning engines involve component troubleshooting, visual inspection (using borescope techniques), fluid contamination checks (oil and fuel) and trimming of the engine. Performing many of these checks on installed engines can reduce the number of spare engines required but affects the aircraft availability rate.

The on-condition maintenance concept differs considerably from the standard engine maintenance concept. Engines maintained under the OCM concept do not have an overhaul time limit that requires engines to be sent to the depot for maximum

operating time. The OCM concept is based upon the identification of life limited components and the tracking of the life consumption of each of these components. Scheduled maintenance actions in the field are not required unless a life limited component reaches its life expiration threshold. This occurrence can be minimized if a life integrity build-up policy is followed during field level and depot level maintenance. Under the OCM concept, the unscheduled maintenance actions control the mean time between engine removal and subsequent manhour and spare part consumption. Since the mandatory periodic inspections are eliminated under the OCM concept, engine monitoring techniques must be available to determine engine condition. The well established oil analysis procedure and borescope techniques provide some of this capability but additional capability is required to monitor in-flight engine or component malfunctions as well as provide a troubleshooting capability during ground operational checks. To the degree that this improved diagnostic capability can fault isolate to defective components, be used to trim the engine, and record performance trending parameters, additional logistic savings are possible. Additional quantification of these requirements has resulted in a need for airborne engine diagnostic equipment.

To demonstrate the capabilities of this type of equipment and further the state-of-the-art in hardware and software development, the US Air Force is currently conducting a formal flight test evaluation of an Engine Health Monitoring System (EHMS) on ten T-38 supersonic aircraft.

The T-38 is a light weight twin-jet advanced trainer which was in continuous production from 1956 to 1972. The Talon as it is designated, is almost identical in structure to the F-5 and is powered by two General Electric J-85-GE-5 turbojet engines, each of which generates 2680 pounds of thrust dry and 3850 with afterburner. The engine health monitoring system is being tested on aircraft stationed at Air Training Command Headquarters, Randolph AFB, Texas.

The system consists of engine sensors, an airframe mounted data processing unit (DPU), and a ground based diagnostic display unit (DDU). The sensors continuously monitor some 24 parameters and input signals to the DPU. Many of the signals were obtained by tapping off of existing sensors, others required the addition of new sensors. The breakout is as follows:

EXISTING SIGNALS	ADDITIONS PICKUP REQUIRED
N - Rotor speed	P_{AMB} - Ambient pressure
W_f - Fuel Flow	P_{TO} - Dynamic boom pressure
A_8 - Nozzle position	P_{53} - Compressure discharge pressure
T_5H - Exhaust gas temperature	P_{55} - Turbine discharge pressure
Engine Oil Pressure	T_{T2} - Total temperature
Anti Ice Switch Position	Engine Oil Tank temperature
Landing Gear Position	Compressor vibration
Utility hydraulic pressure	Inlet Guide Vane/Bleed Value position
Flight Control hydraulic pressure	Throttle angle
26V 400 Hz reference	Fuel Boost Pump pressure
A/C signal return reference	Anti-Ice temperature
28 V DC Power	
EGT Trim	

Signals from these sources are fed into the airframe mounted Data Processing Unit. It is a small (5" x 6" x 10"), light weight (10 pounds), solid state computer. It continuously monitors the input signals from the airframe and each engine. The data is only stored, however, under certain conditions:

a. When any parameter value moves outside previously established performance limits.

b. Upon pilot command,

and/or

c. Under programmed stable flight conditions.

I.E., whenever a parameter limit is exceeded or the pilot desires, the airborne unit will process and record all parameter data as of that moment. An additional record of all data will again be made each time a similar or new condition occurs. In addition, stabilized engine performance data is stored once each flight for pre-takeoff, takeoff, climb, and cruise operating modes. Data collected under these stable conditions is then plotted over time to give an indication of performance degradation. This feature is also used to provide the ground crew with data enabling them to verify with confidence that engines are operating within normal limits and the engine condition is not marginal. Access to this type of data at the flight line can reduce inspection requirements and unwarranted maintenance actions.

The Data Processing Unit is connected to an aircraft status panel which gives an engine and an EHMS go, no-go indication. When an aircraft returns from a flight, the ground crew checks the status panel, if both lights are green, the aircraft can be rapidly turned around for another flight. If there has been an engine problem, which may or may not have been detected by the crew, the Diagnostic Display Unit can be used to troubleshoot the malfunction. The system has not only virtually eliminated power check pad runs to verify pilot write-ups and done away with the all too frequent "cannot duplicate" finding but has reduced troubleshooting time as well.

Should the built-in-test capability of the EHMS indicate a problem with the system itself, the DDU can also be used for troubleshooting purposes.

Whether there has been a problem with the propulsion system or the ground crew is merely collecting the static performance data gathered during the days flights, the data in the airborne memory is transferred to the DDU via a retractable umbilical cord. After this single cord is connected to the aircraft, the transfer is initiated by pressing the data transfer switch on the ground unit. An indicator on the unit indicates when all the data has successfully been transferred. This process is completed in less than five seconds. Any of the data can then be immediately displayed on the portable DDU or carried back to the shop for hard copy printout, transfer to another type of memory unit, or input into a central data bank.

The ground unit is about the size of a brief case (8 1/2" x 12" x 18") and weighs less than 40 pounds. It is powered by self contained batteries and has the capacity to store and present more than 50 data records, normally representing 12 typical flights. It is a non-dedicated unit applicable to any aircraft and can also be used in the test cell or on the trim pad.

During the flight evaluation program, an extensive amount of data is being collected on the ten aircraft and 20 instrumented engines, as well as on a control group of an additional 20 engines. Throughout the test a series of logistic comparisons are being conducted to determine the extent to which such a system can:

a. Fault isolate to module/LRU level

b. Reduce spare engine and piece part requirements

c. Reduce fuel consumption

d. Improve flight safety

e. Mechanize data collection

f. Reduce scheduled engine removals

g. Trend performance data

and in general

h. Increase aircraft availability while reducing logistic support costs by allowing the adoption of a true on-condition maintenance concept.

Results to date indicate that an EHMS system will reduce troubleshooting times as well as increase the basic ability of the maintenance personnel to fault isolate a particular problem. As mentioned earlier, troubleshooting times have been reduced, power check pad runs to verify pilot write-ups have been virtually eliminated, the frequency of "cannot duplicate (CND)" conditions has been greatly reduced, and malfunctions not reported by the aircrews have been detected before they could cause additional damage or adversely affect flight safety. The maintenance personnel particularly like the ability to collect engine parameter data at the time of a malfunction, not only on the defective engine but also on the other engine operating in the same environment. This capability becomes increasingly valuable as the complexity and cost of the engine increase and as we continue to move towards modular and on-condition maintenance concepts.

A direct result of this capability will be a reduction in spare engine and piece part requirements. Although the data base for the T-38 test is not large enough at this time to substantiate and/or quantify these reductions, the system has demonstrated capabilities which indicate that such decreases will occur. The increased troubleshooting capability just discussed will reduce the base repair cycle. Further reductions will result from a reduced requirement for engine test stand runs prior to inputting an engine into maintenance. The data from the last flight will provide the necessary information. A reduction in the base repair cycle will equate to a reduction in the number of spare engines required to support a given flying hour program.

The reductions in piece part requirements result from a decrease in the number of premature engine removals, a decrease in the amount of "trial and error" component replacements, and a decrease in secondary engine damage. By monitoring key performance parameters throughout the flight envelope, the maintenance personnel have more confidence in the actual condition of the engine and remove it only when such an

action is actually required. The result is an increase in the average life extracted from the engine components. The increased troubleshooting and reduced number of CNDs will all but eliminate the all too frequent practice of trying to find the faulty component by trial and error. The net result will be a decrease in the number of units which retest OK at the repair facility and a corresponding decrease in the number of assets tied up in pipeline status. Still other decreases in piece part consumption will result from a reduction in secondary engine damage. This and other USAF test programs have demonstrated an ability to detect relatively minor faults (even though there were no crew write-ups) before they cause significant secondary damage. In many cases there is a shift in one or more of the key engine performance parameters prior to major secondary damage. For example, after a recent flight an EHMS instrumented T-38 indicated compressor and turbine damage. The pilot did not report a problem and when the aircraft returned to base the engine was operating well within performance limits. When the engine was torn down, however, the damage picked up by the EHMS equipment was evident. A small piece of the combustion liner was missing and there was minor secondary damage to the first and second stage turbine buckets. When the piece broke loose and went through the engine, it changed the performance characteristics enough to trip the flags. After it went out the exhaust the engine returned to normal operation. In instances like this, it would be common for additional, larger pieces of the liner to break off and engage the rotating hardware until there is major engine damage and/or the possible loss of an engine or aircraft.

Many other more dramatic examples were encountered in the test of an engine monitoring system on one of our transport type aircraft. In this test, EGT, RPM, PLA, and fuel flow readings were recorded once each flight under stable cruise conditions. This data was then corrected to a selected flight atmospheric base and plotted on individual engine graphs. The first year the test was conducted on a sample of 400 engines. During that time 12 engines, each with interval problems not detected by the crews, were removed before they could cause severe secondary damage. Examples of the damage include combustion chamber liner failure, combustion can burn through, bad fuel nozzles, a hole in an intermediate compressor case, broken stator vanes, and FOD in a compressor. In addition to these 12 engines, 21 others with minor faults were detected and repaired in an installed status, 40 faulty instruments were detected, and "shellouts" were eliminated. A shellout is defined as a piece of the engine coming loose, engaging the rotating hardware, and literally leaving a shell. The 400 monitored engines did not experience a shellout during the entire service test, while the remainder of the fleet sustained 32 shellouts in the first quarter alone. The resultant dollar and safety impacts are obvious.

Another advantage of an EHMS system, particularly important at this time, is its ability to reduce fuel consumption. Some of the areas of reduction have already been discussed; reduced ground run time for troubleshooting and the elimination of test stand runs prior to input into maintenance. Other savings will accrue as a result of reducing the number of functional checks flights (FCFs) required, collecting trend data during normal flight operations, and significantly reducing the fuel requirements for a normal trim operation. The T-38 test has shown that an EHMS system can eliminate a normally required FCF by pinpointing the cause of an otherwise unexplained pilot detected abnormality, by explaining that a flameout was due to operation in a marginal portion of the flight envelope, and by extending the times between scheduled inspections and their subsequent check flights. Other reductions in fuel consumption are enjoyed simply because trend data can be collected during normal flight operations in lieu of periodic runs on the trim pad. Similarly, this type of system has the potential of doing inflight performance checks. Such a check may indicate that an engine needs to be trimmed sometime in the near future. The trim can then be scheduled during programmed non-operational periods. This will eliminate the need for periodic trims at scheduled intervals. Not only will the number of trims be reduced, but whenever a trim is required the EHMS will also permit a reduction in the run time and fuel consumed. To trim an EHMS instrumented aircraft you merely have to connect the DDU with the single umbilical cord discussed earlier. Plugging in one cord will provide a real time digital display of all of the trim parameters. There is no requirement to break into any lines and connect the normal trim equipment, consequently there is no need to remove this equipment, perform all of the standard leak checks, and check run the engine. The T-38 system has saved 1/3 to 1/2 the fuel normally consumed in a typical trim operation.

A side benefit or perhaps the most important aspect of the system for single engine aircraft applications is its ability to improve flight safety. The increased fault isolation capability, the detection of malfunctions not noticed by the aircrews, and the reduction in secondary engine damage all favorably impact flight safety. For example, on a recent flight an EHMS equipped T-38 experienced a stuck nozzle. The pilot and the health monitoring system both reported the nozzle problem but disagreed on its affect. The pilot reported a slight rise in EGT on touch down to 680°C for only a second or two before he shut the engine down. That is not enough of a temperature increase and certainly not for a sufficient period of time to cause any problem. The standard procedure would be to correct the nozzle problem and release the aircraft. The EHMS, however, recorded an EGT of 779°C at a flight level of 27,000 feet for a duration of 41 seconds. That temperature for that length of time indicated severe damage. The engine was pulled and a hot section inspection revealed that the turbine had been virtually melted. Another flight may well have resulted in an incident or accident.

All of the features discussed thus far are important but the single most vital aspect of an EHMS may be its ability to mechanize engine data collection. The improved integrity of the record keeping of time and/or cycle limited components and the ability to trend performance degradation on EHMS instrumented engines will improve our capability to forecast engine overhaul and MISTR (Management of Items Subject to Repair) requirements. This near real time data will also improve the accuracy of projections of spare parts consumption. We know from experience on other engines what is normally required in parts replacement during periodic inspections (P/Es). An example is the replacement of most (90%) chambers and some (40-50%) of the first stage turbine nozzles on J79 engines at the 600 hour P/E. Engines maintained under an OCM concept, however, will not have a P/E. Modules/assemblies will be replaced on-condition thus forcing the parts replacement identification (and trending) to be on a real time, event-by-event basis. With an EHMS we will be able to record what engine performance and key parameter values were for certain parts being replaced, thus providing an auditable trail of condition versus support costs. An additional benefit of an automated data acquisition capability is the early identification of engine failure trends to aid in the selection of appropriate component improvement program (CIP) tasks.

The net result of all of this should be an increased ability to keep the engine installed; i.e., reduce engine removals. This system will not only reduce premature removals as discussed earlier but will extend the time between scheduled removals as well. In both the T-38 test and the experiment on transport type aircraft, favorable parameter trends have been the basis for authorized overflys of scheduled engine removals. This is an indication of the progress toward and subsequent reality of true on-condition maintenance.

In the final analysis, the major contribution of the EHMS flight test may be in the area of capabilities demonstrated and lessons learned. It has and will surely continue to aid in the definition of appropriate systems for fighter aircraft as well as define engine condition monitoring policies and procedures applicable to all aircraft.

DISCUSSION

Anonymous

Mr Tönskötter just in the lecture before mentioned something about the difficulties in finding local defects in the engine. What has been your experience in the necessity to install a diversity of sensors, e.g. to measure temperature at turbine inlet to find local defects such as fuel nozzle malfunctions or pieces broken out of the combustor, or pieces broken out of the turbine nozzle?

Author's Reply

How much of that can be done is certainly subject to discussion and it varies from engine to engine and from application to application. We believe that to determine how much you can do and should do in a particular application, that you really need to buy some hardware, fit it on an airplane, fly up and find out.

Anonymous

How do you find the position of your people checking your aircraft after having introduced your monitoring system? I can visualize that it is possible that your maintenance people will start highly depending on the system and may become more careless in their further portion of the inspection.

Author's Reply

I do not know the results yet. That is one of the things we are looking at. I think that perhaps the problem is not the other inspections per se. It is the inspections that will be eliminated altogether by this type of equipment. That should you choose to fly the aircraft without the equipment operating properly you will have to go back to your old maintenance practices and then you may have a problem.

Anonymous

You may have a problem then when you need to redeploy your aircraft in an area where you do not have the monitoring system available. That may limit your flexibility in redeployment.

Author's Reply

That's true, that is why the reliability of this type of equipment is so critical.

H.I.H.Saravanamuttoo

When you are looking for parameter limit overcrossing are you using manufacturers average base line data or individual engine base line data?

Author's Reply

We are using individual engine data. We have a signature for each individual engine.

ROUND TABLE DISCUSSION

Speakers:

A.MIHAIL, Bureau Veritas, Paris, France
G.P.SALLEE, United Aircraft Corp., East Hartford, USA
M.J.GUEST, RAF, MOD, London, UK
R.SMYTH, VFW-Fokker, Bremen, Germany
J.C.RIPOLL, CEPr, Saclay, France

ROUND TABLE DISCUSSION

Monsieur A.Mihail: La nombreuse assistance, la haute teneur des communications et le nombre des questions posées prouvent que le choix du thème de ces journées, fait par le Président et la Commission adéquate, était non seulement pleinement justifié, mais nécessaire.

Vous connaissez probablement tous le boutade du marchand de parachutes disant "si le parachute ne s'ouvre pas je le remplace gratuitement". Le problème posé par la sécurité de fonctionnement du parachute est aussi le problème de sa fiabilité car en fait, et en règle générale, il ne s'agit pas seulement de savoir ce que rapporte la fiabilité, mais plutôt ce que coûte le manque de fiabilité et ses conséquences.

Comme l'a si bien fait remarquer Monsieur Koff (par personne interposée), la fiabilité est une oeuvre de création continue: un moteur à réaction est un ensemble très complexe où se trouvent rassemblés à la fois des phénomènes aéro-thermodynamiques, chimiques, physiques, électriques, etc. . . On ne peut, donc, vouloir faire ou rendre fiable un moteur qui n'a pas été "pensé" lors de sa conception en fonction de cette donnée, et de cela on doit tenir compte à tous les stages depuis le projet jusqu'à, et y compris, l'exploitation, en passant par la production, la mise en service, la réparation, etc. . . La fiabilité d'un réacteur doit être prise en compte lors de la conception au même titre que ses performances. Il s'agit donc, en premier lieu, de créer un état d'esprit car la fiabilité, dans le cadre d'un projet aussi complexe qu'un moteur, est avant tout cela.

Il est bien connu que pour rendre un moteur réellement compétitif en exploitation, le Constructeur doit admettre une dépense équivalente à celle consentie pour sa certification. On peut certes améliorer la fiabilité du moteur en exploitation, mais le coût de cette amélioration sera d'autant plus élevé que l'importance accordée à la fiabilité au stade du projet aura été plus faible.

En d'autres termes, considérer la fiabilité comme un facteur d'égale importance que les performances est devenu une nécessité absolue pour les moteurs civils et dans le contexte actuel il en sera de même, vraisemblablement, pour les moteurs militaires.

* * * * * *

On a beaucoup parlé, durant ces deux journées, des différents aspects de la fiabilité du moteur et de certains de ses constituants, ainsi que des coûts. Le remarquable exposé du Général Giorgieri et du Colonel Facca se passe de commentaire.

Mais sur un autre plan, plus abstrait celui-là, je vous rappelle que M. Slatford parlait de taux de défaillance de 10^8 pour l'ensemble du groupe propulseur, autrement dit, y compris tous ces accessoires et ces instruments de conduite et de contrôle (dont vous avez noté l'importance pour les programmes "On Condition"). Or, c'est justement là, qu'à l'heure actuelle se pose avec le plus d'acuité le problème du manque de fiabilité.

La tenue en service des accessoires et des systèmes de contrôle moteur constitue présentement le point faible du groupe. M. Rennesson l'a bien montré dans un de ces exemples.

A une de mes questions, M.Demarteau – de la K.L.M. – a confirmé la mauvaise fiabilité du système de surveillance AIDS lui-même. Quand, par ailleurs, la surveillance continue des paramètres de fonctionnement du moteur a éte mise en route, cela a eu pour effet, en premier lieu, de démontrer le mauvais fonctionnement des instruments de contrôle et de conduite moteur, qui eux, et non pas le moteur lui-même, étaient à l'origine de nombreuses déposes prématurées, voire d'arrêt en vol.

En raison de la nature et de l'origine des équipements, je ne voudrais pas créer d'incident "diplomatique", mais la vérité oblige à dire qu'il n'y a pas, par exemple, de système de détection incendie ayan une fiabilité satisfaisante du moins pour les avions commerciaux. Les Constructeurs ont été obligés de la doubler sur les appareils de la dernière génération (DC.10, B.727, B.747, AIRBUS, etc. .). Cette apparent redondance pose d'ailleurs d'autres problèmes, mais cela est une autre question. Quand on sait qu'une détection incendie (et les fausses détections représentent bien plus de la moitié du total lorsque l'on utilise le système simple) entraîne, outre l'arrêt du moteur, la vidange éventuelle (suivant la phase de vol) de plusieurs tonnes de carburant, le tout suivi du retour à la base de départ, il y a là matière à réflexion pour ceux qui font des recherches sur la fiabilité ou des études de coût/efficacité.

Relever avec fidélité une température c'est déjà difficile, mais quand de plus, la fiabilité du système de mesure est faible, voir douteuse, vous concevez les dépenses en main-d'oeuvre et surtout en pièces que les résultats d'une telle mesure incorrecte peuvent entraîner.

C'est pourquoi, Messieurs, je pense que les problèmes de fiabilité des accessoires et des systèmes de conduite et contrôle moteur mériteraient qu'on leur consacre, à eux seuls, une session spéciale de votre assemblée.

Mr G.P.Salle: We had a presentation yesterday on FOD – Foreign Object Damage – We have seen that a fighter inlet that sits as high as the Electra will not pick up large objects from the ground into the inlet. But I do not believe that the experience applies to a large high by-pass engine that is pumping fourteen or fifteen hundred pounds of air per second. Those results are quite different. These large engines do generate a large vortex and do pick up sand and even larger particles.

In looking over what has been said in this meeting, I think that the commercial airlines have shown an ability to tell us in detail those parts that have been breaking and their dissatisfaction with the state of the reliability of the new technology engines. The military also is saying essentially the same thing. I see emphasis on changing the policies and

procedures in the design and development of engines to try to put more emphasis on reliability than we have seen before. But it seems strange to me that we know so little about: (1) how the engines are performing today in terms of performance, and (2) how much it costs to maintain engines. While we know that there are many causes for engine removal that are similar between nations, we have yet to study the significant differences in engine reliability and reliability experience between nations.

As a case and point, I go back to commercial airlines. I find in my studies at Pratt & Whitney that the mean time between shop visit of the JT8D engine can vary from 2500 hrs to 3800 hrs, depending on which airline you want to talk about. The same variability in shop visit rates exists in all other engine models I have looked at. Further, the airline that seems to do the best on one engine does not necessarily do so well on another. There are gross differences in engine build/repair standards and in how much money is put into or invested in repairing engines as I move from one airline to another. From the discussions we had with the military in Working Group No.8, the same thing is true if I look at the same engine in different nations. There is a great deal of variability in what is actually done in a normal engine repair process. I believe we are just starting the study of reliability rather than ending it.

Group Captain M.J.Guest: I find it very difficult indeed to comment on the feast of information that we have enjoyed for the last two days. However, as a representative of one of the NATO Air Forces and thus as a military customer two things have interested me particularly. Both concerned money, and I would remind you of the trends that we have seen in some of the papers, which show how steeply component replacement costs have been rising.

First, we heard yesterday of the civil thinking on the risks of failure that should be tolerated with civil engines. Compared with civil aircraft there is generally a much higher risk level associated with military aircraft in peace time for a whole range of reasons; not the least of which may be the exuberance of fighter pilots mentioned this morning. Are we therefore applying unnecessarily low and therefore expensive levels of acceptable failure probability in our military engines when we are deciding on the lives of expensive components? Perhaps an increase in acceptable failure level might result in an insignificant change in the overall risk level but still give a valuable increase in life. Although some military customers may already be exploiting this thought, we in the UK are becoming very interested in this field of risk assessment, certainly from the customer's point of view.

Second, because of the extreme variation that we observe between two identical engines in identical aircraft doing an apparently identical job, we feel intuitively that we ought to gain financially by lifing each military engine component on its condition. To do so, we would need some form of continuous monitoring of engine parameters such as r.p.m. and temperature on each engine followed by some form of automated condition assessment.

However we are having difficulty at this moment in finding any concrete evidence to use in convincing our financial masters that buying hardware to monitor condition in this way will save money in the future. Although I am by no means an expert, the missing link seems to me to be that we have yet to demonstrate that we can adequately assess the actual condition of components from records of their in-service use, and if anyone has such evidence we would be delighted to see it.

In conclusion, may I add my thanks to the Panel and to all the presenters of PEP for the obvious amount of very hard work that has gone into making this symposium a success.

Mr R.Smyth: The main point of interest in the papers presented and in the discussions of this PEP-meeting on Engine Reliability has been the engine as a vital hardware unit which has a specific function to fulfill and also costs money during usage. Thus there are three categories which engine reliability can be divided into:

(a) mechanical integrity,
(b) function of the engine (performance, deterioration),
(c) economics.

Mechanical Integrity

The mechanical integrity has been the main subject of discussion as this is directly coupled to reliability. Mechanical failures must be avoided during usage. The way an engine is expected to be put to use in service and the actual type of usage it then experiences are very important factors governing the mechanical integrity of a given engine in a given installation.

A number of papers have been presented showing what measures can be taken to build a high standard of durability into an engine including its subsystems. There have been very interesting presentations showing what can be done in the design phase and how engine durability can be tested in short time. Appropriate test procedures with actual hardware can be applied to find out whether the engine is fulfilling the durability demand from it. This assumes a specific method of usage.

When the engine goes into service actual usage must be monitored and compared with the assumptions laid down to demonstrate the standard of durability demanded during development. Various papers on the usage of engine recorders, e.g. life history recorders, have been presented. Experience of a commercial airline with engine diagnostics has also been discussed.

Concerning the life usage of critical components there were discussions on the evaluation of low cycle fatigue (LCF) and of turbine creep life. There is sufficient know-how available to build up useful mathematical models for calculating engine life usage in real time in flight.

The analysis of service usage to monitor the rate of life usage compared to predictions during the development phase must include engine handling. Different modes of engine handling at elevated temperatures can lead to great differences in engine life usage.

Investigations showing the behaviour of engines after the failure of critical parts are necessary for early failure recognition and fault analysis. There have been very interesting papers presented showing actual test results with a failed turbine blade and clogged fuel spray nozzles.

Related to engine usage in service foreign object damage (FOD) can cause serious engine damage leading to substantial performance losses, critical failures and expensive engine overhauls. The RAF film for Dr Fod was very informative in showing how simple precautionary measures can be in avoiding FOD.

Performance Reliability

Besides the mechanical integrity engine reliability must also include the reliability to perform as expected. Engine usage leads to different grades of performance deterioration. The subject of performance restoration after overhaul has been discussed during this meeting. The question of when to overhaul an engine to obtain a certain amount of performance restoration for a given cost is still open. Further investigations in this field are necessary. The balance between maintenance costs and performance restoration is an important factor determining performance reliability in service.

Economics

As already shown above reliability costs money. This is the economical aspect of reliability. During this meeting we have learnt that mechanical failures can involve big sums of money, in the order of hundreds of thousands of dollars. Military operators often tend to overlook these immense cost aspects. It is very useful to talk to colleagues in the commercial airline business from time to time in order to get a feeling for the cost factors involved when reliability of an engine is down.

What we still lack are comprehensive costing models for engine usage. These models must take account of differences between engines and deliver a basis for realistic comparisons of the effects of different measures on engine economics and total money spending. The concept of engine life cycle cost (LCC) is gaining increased attention in the USA. This appears to be a reasonable starting point for analysing the economic aspects of engine reliability.

Monsieur J.C.Ripoll: Je voudrais maintenant conclure assez brièvement, après les quatre exposés que ces Messieurs ont bien voulu nous fair. Je crois que nous avons fait une bonne synthèse de la situation et je ne m'étendrai pas longuement. Je dirais, d'un point de vue philosphique, que pendant assez longtemps, l'aéronautique a certainement pensé plus "materiel", c'est-à-dire vitesse, sophistication, performance, que "fonction", c'est-à-dire, transport de personnes, confort, régularité, profit, et les équivalents pour les militaires. Pour illustrer ceci, j'ai trouvé une phrase que je vais essayer de lire en anglais dans un document émis par un groupe d'ingénieurs de Wright-Patterson: "The new generation of gas turbine engines is thoroughbred, with all the idiosyncrasies of the thoroughbred breed".

Mais désormais, avec la forte pression qu'exerce sur nous l'écologie, la croissance des coûts, (du personnel comme du pétrole), les difficulties de l'économie internationale, je crois que cette conception doit évoluer profondément et, la fiabilité des propulseurs est un élément important de ce problème.

Ce que j'ai retenu, pour ma part, de cette réunion, c'est que toutes les procédures doivent être revues, en particulier avec cet aspect fiabilité en vue. Au niveau de la conception, par des relations plus intimes entre le concepteur et l'utilisateur, l'utilisation de méthodes de travail comme le ENSIP comme "l'approche-système" au niveau du développement, en appliquant par exemple, cette notion d'essais accélérés; au niveau de la certification et de l'homologation, par une meilleure définition des essais; au niveau de l'évaluation, des coûts et performances – on n'a peut-être pas assez parlé du concept de life-cycle-cost et du programme "APSI cost" . . .; au niveau de la maintenance, on a peut être pas assez parlé de "poursuivre l'effort du Maintenance Steering Group" pour imposer une méthodologie; au niveau de la surveillance, en faisant un peu plus référence à une sorte de médecine thermomécanique.

En tant que Président du Comité de ce meeting, je remercie tous les auteurs qui ont contribué par d'excellentes communications sur ce sujet. Je remercie aussi nos quatre orateurs à côté de mois, pour leur contribution synthétique, je remercie évidemment l'assistance d'avoir bien voulu nous honorer en grand nombre jusqu'au dernier moment et d'avoir réellement animé cette réunion; je veux aussi remercier nos interprètes qui ont donné, et je parle avec une certaine expérience, car j'écoute toujours les deux versions, française et anglaise, elles nous ont donné une *très* bonne interprétation.

Je pense maintenant que le Dr Winterfeld, Président de notre Panel Energétique et Propulsion, va vous dire quelques mots sur nos prochaines réunions, et vous verrez je pense, que notre Panel fait de son mieux pour répondre aux préoccupations qui ont été spécialement mises en valeur par ces Messieurs.

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The meeting was organized to review and discuss engine reliability from four aspects:

1 • the reliability of current civil and military engines as experienced by the users,
2 • civil and military authorities' plans to promote improved reliability in future engines,
3 • what manufacturers are doing to improve reliability through design and testing programs,
4 • the role that engine health monitoring and diagnostics is taking in minimizing the impact of engine unreliability for both civil and military users.

High Engine performance, achieved step by step from one engine generation to the next has to be paid for with higher costs both of the original engine and the maintenance due to the increased complexity. Today, economic forces have produced the need for both users and manufacturers to re-evaluate their priorities on performance and engine reliability.

The meeting gave insight to major causes for engine unreliability (problems arising repeatedly and with every user, difference between engine usage and design basis, dependency on auxiliary parts or components of the engine) and showed the need for comprehensive data feedback to manufacturers.

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